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AIRWORTHINESS AND FLIGHT CHARACTERISTICS EVALUATION C-12A AIRCRAFT

FINAL REPORT

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UNITED STATES ARMY AVIATION ENGINEERING FLIGHT ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA 93523

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

The United States Army Aviation Engineering Flight Activity conducted an airworthiness and flight characteristics evaluation of a C-12A aircraft, serial number 73-22250, from 25 October 1975 through 14 February 1976. The aircraft was tested at Edwards Air Force Base (field elevation 2302 feet), Paso Robles (field elevation 836 feet), and Lake Tahoe (field elevation 6262 feet), California. During the evaluation 71 flights totaling 68.75 productive flight hours were → next page

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20. Abstract

cont

conducted. Performance and handling qualities of the C-12A were evaluated under a variety of operating conditions with emphasis on operation in the normal mission configuration near the maximum gross weight of 12,500 pounds. The test aircraft was evaluated against the requirements of Federal Aviation Regulation Part 23, the Beech Aircraft Corporation prime item development specification, and military specification MIL-F-8785B(ASG) to assist in determining operational mission capabilities. Two handling qualities deficiencies were identified. These were the main landing gear wheel lockup tendency which occurred when applying brakes during landing, and the lack of adequate stall warning above 20,000 feet pressure altitude. Twenty shortcomings were noted including four stability and control shortcomings, two lighting system shortcomings, and 14 reliability and maintainability shortcomings. The C-12A failed to meet the single-engine service ceiling, dual-engine cruise ceiling, and the 30,000-foot altitude cruise airspeed guarantees. Two enhancing features were the location of the landing light switches and the rudder boost, which greatly reduced pilot workload during asymmetric power conditions.

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DRSAV-EQ 23 May 77

SUBJECT: Directorate for Research, Development and Engineering Position on the Conclusions and Recommendations of the Final Report on USAAEFA Project No. 75-08, Airworthiness and Flight Characteristics Evaluation C-12A Aircraft, dated October 1976

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1. The Directorate for Research, Development and Engineering position on USAAEFA's conclusions and recommendations are provided herein. Paragraph numbers from the subject report are provided for reference.

a. Para 83a. The C-12A aircraft as tested met all contract guarantees, with three exceptions: (1) dual engine c uise ceiling, (2) single engine service ceiling, and (3) dual engine cruise speed at heavy gross weight and high altitude. However, it is also recognized that the slight additional power required to meet the three guarantees that were missed can be obtained from the current engines while maintaining an adequate temperature margin. Therefore, action has been taken to incorporate changes in the specification and the operators manual which will allow this additional power to be used. This results in the C-12A meeting all the performance guarantees.

b. Para 84a. The main landing gear wheel lockup tendency during landings using brakes is a deficiency. A recommendation to equip all C-12's with an anti-skid wheel brake system was made to HQS, DARCOM, and the Department of the Army in July 1976. To date, this improvement has not been funded.

c. Para 84b. The lack of adequate stall warning at pressure altitudes above 20,000 feet was a deficiency on the test aircraft. Subsequent investigations have revealed that a change in the nominal stall warning computer excitation value will result in adequate stall warning for all C-12's. Several other aircraft were checked during production acceptance test flights and the only known case of this deficiency is that experienced with the test aircraft. However, action has been taken to provide revised excitation value tolerances for the stall warning computer, and to check all aircraft already fielded for proper stall warning operation and repair as required.

d. Para 85a, b, c and s. The shortcomings outlined in the referenced paragraphs deal with poor long term trimability, easily excited dutch-roll oscillations, lightening of elevator control forces and awkward cockpit ingress and egress. These items are considered minor in nature and the effort required to overcome them is not justified, therefore, no action is being taken on these items.

e. Para 85d. The unsatisfactory yaw damper operation in turbulent air was due to a malfunctioning yaw damper on the test aircraft. No action is being taken on this item.

f. Para 85e. The inability to dim the warning, caution and advisory panel lights with the instrument indirect rheostat on is not considered a shortcoming. The lighting system was designed in this manner to assure that the warning, caution, and advisory panel lights would be clearly visible when the instrument indirect lights were on. The design is, therefore, correct as currently configured.

g. Para 85f. The main spar presents an obstacle in the cabin aisle which cannot be readily seen when moving about the cabin in flight at night. Action will be pursued to make the spar more visible.

h. Para 85g. The poly flow tubing used to sense bleed air failures is designed to rupture when exposed to elevated temperatures, thus its failure after vibrating loose in flight and being exposed to excessive heat from adjacent aircraft components is normal. The failure of the tubing attachments is a maintenance problem which has only occurred on the test aircraft, therefore, no action is being taken on this item.

i. Para 85h. The propeller proximity switch has been removed from the system thus eliminating this shortcoming.

j. Para 85i, k, l, g and r. The shortcomings outlined in the referenced paragraphs are routine maintenance problems which will be considered along with future reports of similar failures on other fielded C-12 aircraft in determining if a change in configuration or maintenance procedure is required.

k. Para 85j. The cabin door seal pressure line tearing loose from the door seal has been experienced on several aircraft and action will be pursued to alleviate this problem.

l. Para 85m. An improvement to throttle friction has been made in later production aircraft and retrofit kits will be installed in all in-field aircraft.

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m. Para 85n. The fuel tank access plates for later production aircraft and all spares have been changed from a sheet metal to a cast design which provides better seal retention and eliminates this shortcoming.

n. Para 85o. The "working" rivets in the test aircraft were the result of improper rivet installation. The condition has been corrected on the test aircraft which was the first production model, and on all other early production aircraft which also experienced this problem.

o. Para 85p. A capacitor has been added to the ice vane annunciator system to eliminate the intermittent illumination of master warning and master caution lights.

p. Para 85t. Action is being pursued to provide instructions in the operators manual to drain the toilet prior to conducting training or other flights in which operation at or less than zero g is expected. A design change would be costly and is not warranted.

q. Para 90, 91b, 91c, 92a and 92b. The recommended addition of the warnings, cautions and notes as outlined in the paragraphs referenced are being incorporated in the operators manual.

2. The C-12A aircraft is considered a fully qualified aircraft in that the required FAA type certificate has been issued and the results of these tests have shown that the additional military airworthiness requirements contained in the Prime Item Development Specification (beyond the minimum FAA requirements) have been met. Incorporation of an anti-skid wheel brake system would greatly enhance the use of the C-12.

FOR THE COMMANDER:

Walter A. Ratcliff

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Colonel, GS
Director of Research,
Development and Engineering

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Same as the AAEFA C-12A Aircraft
Report Project No. 75-08

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INTRODUCTION

BACKGROUND

1. The C-12A is a fixed wing utility aircraft manufactured by Beech Aircraft Corporation (BAC) and procured under United States Army Aviation Systems Command (AVSCOM) contract DAAJ01-75-C-0941 dated 13 August 1974. A United States Air Force version of the same aircraft, the C-12A(AF), was also procured under the same contract and specification. The mission of this aircraft is to be a utility transport for passengers and/or cargo. The C-12A is a derivative of the Beech Model 200 aircraft and was type certified in the normal category of Federal Air Regulation (FAR) Part 23 (ref 1, app A) of the Federal Aviation Administration (FAA). The United States Army Aviation Engineering Flight Activity (USAAEFA) was directed by AVSCOM to perform an airworthiness and flight characteristics (A&FC) evaluation (ref 2).

TEST OBJECTIVE

2. The objective of this test was to quantitatively and qualitatively evaluate the C-12A aircraft to determine the degree of compliance with the following:

a. Performance guarantees, as specified in contract DAAJ01-75-C-0941.

b. Stability and control requirements of the BAC prime item development specification (PIDS) (ref 3, app A).

DESCRIPTION

3. The test aircraft was a C-12A, serial number 73-22250, powered by two United Aircraft of Canada, Ltd (UACL) PT6A-38 turboprop engines. This aircraft is the military version of the BAC Super KingAir Model 200 pressurized all-weather executive transport. The pilot and copilot are seated side by side with dual flight controls. The tricycle landing gear is retractable, with dual wheels on each main gear. The control system is fully reversible. A pneumatic rudder boost is installed to help compensate for asymmetrical thrust and a yaw damper system is provided to improve directional stability. A detailed description of the C-12A is contained in the BAC PIDS. Appendix B contains a further description of the test aircraft.

TEST SCOPE

4. The A&FC tests were conducted at Edwards Air Force Base (field elevation 2302 feet), Paso Robles (field elevation 836 feet), and Lake Tahoe (field elevation 6262 feet), California, from 25 October 1975 through 14 February 1976. During the test program 71 flights were conducted for a total of 123.7 hours, of which 68.75 hours were productive. The first production C-12A aircraft, serial number 73-22250, was used throughout the test program. The C-12A was evaluated to determine its overall performance capabilities, handling qualities characteristics, specification compliance, and to verify contract guarantees. The airplane configurations are presented in table 1 and the test conditions are shown in tables 2 and 3. Flight restrictions and operating limitations applicable to this evaluation are as approved by the FAA and contained in the BAC operator's manual (ref 4, app A).

Table 1. Airplane Test Configurations.

Configuration	Gear Position	Flap Setting (% full down)	Power Setting	Propeller Speed (rpm)
Takeoff (TO)	Down	Zero and 40	TO[1]	2000
Climb (CL)	Up	Zero	MCP[2]	2000
Cruise (CR)	Up	Zero	PLF[3]	1800
Power approach (PA)	Down	40	PNA[4]	2000
Landing (L)	Down	[5]100	Flight-idle	NA
Glide (G)	Up	Zero	Flight-idle	NA
Waveoff (WO)	Down	100	TO	2000

[1]Takeoff power: Power at maximum torque (1970 ft-lb).
[2]Maximum continuous power in climb: Power at 1920 ft-lb torque.
[3]Power for level flight: Power required to maintain level flight.
[4]Power for normal approach: Power required to maintain 3-degree descent.
[5]100 percent flaps = 33 degrees.

Table 2. Performance Test Conditions.

Test	Pressure Altitude (ft)	Airspeed	Gross Weight (lb)	Center-of-Gravity Location	Configuration
Level flight	Sea level[1]	$1.1V_S$[3] to V_H[4]	11,280 and 12,000	Forward[5]	CR
	[2]5000				
	[1]10,000				
	20,000				
	25,000				
Sawtooth climbs	Sea level[1]	$1.1V_S$ to V_H	12,000	Forward	CL
	[2]15,000				
	25,000				
Takeoff and landing	5000	$1.2V_S$	12,000	Forward	TO and L
	Sea level and 6000	$1.1V_S$ to $1.4V_S$			
Airspeed calibration	5000	$1.1V_S$ to V_H	10,000	Forward	CR and WO
Continuous climb	Near surface to maximum altitude	$V_{max\ R/C}$[6]	12,000	Forward	CL
Stalls	10,000	V_S	11,280 and 12,500	Aft	Dual-engine TO, CR, L, and PA. Single-engine CR and CL.

[1]Dual and single-engine.
[2]Single-engine only.
[3]V_S: Stall airspeed.
[4]V_H: Maximum airspeed for level flight.
[5]Limited testing conducted at aft center of gravity (cg) to determine effect of cg on level flight performance.
[6]$V_{MAX\ R/C}$: Airspeed for maximum rate of climb (provided by contractor).

Table 3. Handling Qualities Test Conditions.

Test[1]	Trim Airspeed	Test Gross Weight (lb)	Center-of-Gravity Location	Configuration
Static longitudinal stability	$1.4V_S$	12,500	Fwd and aft[2]	PA
	V_{CR}[3]			CR
Static lateral-directional stability	$1.4V_S$	12,500	Aft	PA
	V_{CR}			CR
Maneuvering stability	130 KCAS	12,500	Aft	PA and CR
	180 KCAS			
	230 KCAS			
Dynamic longitudinal stability	$1.4V_S$	12,500	Aft	PA
	V_{CR}			CR
Dynamic lateral-directional stability	$1.4V_S$	12,500	Aft	PA
	V_{CR}			CR
Trim characteristics	[4]Per 23.161b and 23.161c(6)	12,500	Aft	CR
	[4]Per 23.161c(1)	12.500	Aft	CL
	[4]Per 23.161c(4)	12,500	Fwd	PA
	[4]Per 23.161c(5)	11,279	Fwd	PA
	[4]Per 23.161d	12,000	Aft	Single-engine CR
Longitudinal control	[4]As required by 23.145b and 23.145c	12,500	Fwd and aft	[4]As required by 23.145b and 23.145c
		11,280	Fwd	
Roll performance	120 to 180 KCAS	[5]12,500	Aft	CR and TO
Single-engine characteristics	V_{MC}[6]	12,500	Aft	Gear up, flaps 40 percent

[1]Test altitude was 10,000 feet.
[2]Center-of-gravity locations:
Fwd - Fuselage station (FS) 183.4 for gross weights above 11,280 pounds; FS 181 for gross weights of 11,280 pounds or less.
Aft - FS 196.4.
[3]V_{CR}: Recommended cruise airspeed.
[4]FAR Part 23, para(s) 23.145 and 23.161.
[5]With maximum fuel in wing tanks.
[6]V_{MC}: Airspeed for minimum control.

TEST METHODOLOGY

5. Established flight test techniques and data reduction procedures were used during this test program (refs 5 through 10, app A). The test methods are described briefly in the Results and Discussion section of this report. Flight test data were hand-recorded from test instrumentation on the pilot, copilot, and engineer panels and automatically recorded on magnetic tape. A detailed list of the test instrumentation is contained in appendix C. Test techniques (other than the standard techniques described in appropriate references), weight and balance, and data reduction techniques are contained in appendix D. Noise level data are presented in appendix E. A Handling Qualities Rating Scale (HQRS) (app F) was used to augment pilot comments relative to the aircraft handling qualities. Deficiencies and shortcomings are in accordance with the definitions presented in Army Regulation 70-10.

RESULTS AND DISCUSSION

GENERAL

6. Performance and handling qualities of the C-12A aircraft were evaluated under a variety of operating conditions with emphasis on operation in the normal mission configuration near the maximum gross weight of 12,500 pounds. The test aircraft was evaluated against the requirements of FAR Part 23, the BAC PIDS, and military specification MIL-F-8785B(ASG) (ref 11, app A) to assist in determining operational mission capabilities. Two handling qualitics deficiencies were identified. These were the main landing gear wheel lockup tendency which occurred when applying brakes during landing and the lack of adequate stall warning above 20,000 feet pressure altitude (Hp). Twenty shortcomings were noted including four stability and control shortcomings, two lighting system shortcomings, and 14 reliability and maintainability shortcomings. The C-12A failed to meet the single-engine service ceiling, dual-engine cruise ceiling, and cruise airspeed at 30,000 feet altitude guarantees. Two enhancing features were the location of the landing light switches and the rudder boost system, which greatly reduced pilot workload during asymmetric power conditions.

PERFORMANCE

General

7. The performance characteristics of the C-12A aircraft were evaluated under various operating conditions, with emphasis on operation in the normal mission configuration near the maximum gross weight of 12,500 pounds at the forward cg limit of fuselage station (FS) 185.0. The C-12A failed to meet the single-engine service ceiling, dual-engine cruise ceiling, and the 30,000-foot altitude cruise airspeed contract guarantee performance. The shaft horsepower (shp) available, fuel flow rate, and net thrust of a PT-6A-38 specification engine were provided by an engine computer program furnished by UACL. The propeller efficiency chart was furnished by BAC and the installation losses of the test aircraft were obtained from the BAC PIDS. All performance guarantees were calculated using these values.

Takeoff and Landing Performance

8. Takeoff tests were conducted to determine the operational technique and takeoff ground and air distance to clear a 50-foot obstacle and to check the contract guarantee performance. All takeoffs were made from a hard, dry, paved level runway. A normal takeoff technique was used throughout the tests (flaps up, holding the brakes, and stabilizing at takeoff power on both engines prior to starting the ground roll). Test conditions are presented in table 2 and the test results are summarized in figure 1, appendix G. Test data are shown in figures 2 through 5. Initially, the contractor's recommended rotation speed of 110 knots indicated

airspeed (KIAS) was used. This rotation speed proved to be impractical, in that positive effort was required to prevent the nose wheel from lifting off at a much lower airspeed. Subsequently, several different rotation speeds were evaluated and 95 KIAS was selected as optimum. The C-12A accelerates very rapidly immediately after lift-off when a constant pitch attitude is maintained. Therefore, to take advantage of this characteristic, the tests were flown holding a constant pitch attitude, rather than trying to maintain a specific airspeed prior to reaching 50 feet. Using this technique, airspeed at 50 feet exceeded the $1.3V_S$ minimum specified in FAR Part 23. The guaranteed takeoff performance distance of 2820 feet over a 50-foot obstacle, on a sea level, standard day at a gross weight of 12,000 pounds, was met with a 20-foot margin.

9. During this test program a short field takeoff technique was developed. The short field technique differed from the normal takeoff technique in that flaps were used (40 percent) and rotation was executed at V_{MC} (87 KIAS). Test results presented in figures 6 through 9, appendix G, indicate that compared to the standard technique, the takeoff ground distance was decreased by 9 and 16 percent at sea level and 6000 feet, respectively. The total distance to 50 feet was decreased by 21 percent at sea level and 29 percent at 6000 feet. Typical time histories of short field technique takeoffs are presented in figures 10 and 11.

10. Power-on landing tests were conducted to verify the contract guaranteed performance and determine the ground roll distance without the use of reverse thrust. All landings were made on a hard, dry, paved level runway. Test results are summarized in figure 12, appendix G, and test data are presented in figures 13 and 14. During the conduct of these tests, the application of moderate braking after touchdown, immediately after flap retraction, resulted in the outboard wheels locking up and blowing tires on several occasions. This outboard wheel lockup tendency could not be readily discerned or anticipated in the cockpit until after the tires were blown. A total of five tires were blown and an additional four changed because of flat spots. The wheel lockup tendency was a deficiency and an Equipment Performance Report (EPR) (ref 16, app A) was submitted. Consideration should be given to incorporating an antiskid wheel brake system on the C-12A. If an antiskid wheel brake system is incorporated, additional testing should be conducted to determine the effect on landing performance. The landing performance of the C-12A aircraft met the contract guarantee of 2514 feet.

11. Short field landing capability tests were conducted. Due to the wheel lockup deficiency uncovered during landing tests (para 10 above), the short field technique was modified. The short field landing technique differed from the normal landing technique by the use of maximum reverse thrust, after touchdown, which was maintained until decelerating to 40 knots estimated ground speed. Due to the probability of propeller erosion and low effectiveness, maximum reverse thrust was not recommended to be used at airspeeds below 40 knots. To reduce the probability of blowout of the main landing gear tires, brakes were not applied until the aircraft had decelerated to approximately 40 knots estimated ground speed. This technique was recommended by BAC. Test results are presented in figures 15 through 18, appendix G. The short field landing performance was

compared with normal landing performance as summarized in figure 12. Typical time histories of landings are presented in figures 19 and 20. The ground distance required using the short field technique was not appreciably changed from that required using the normal technique and maximum braking. During these tests a preselected approach airspeed was maintained until reaching 50 feet, then a specific touchdown speed was achieved. Due to the aircraft's tendency to float in ground effect (IGE), shorter landing distances could be achieved by aiming for a particular touchdown spot and holding airspeed constant throughout the entire approach.

Climb Performance

Sawtooth Climb:

12. Dual and single-engine climb performance was evaluated at the conditions shown in table 2, using the sawtooth-climb method of test. All dual-engine climb tests were conducted with both engines operating at maximum continuous power (MCP). All single-engine climb tests were conducted with the left engine shut down and the propeller feathered, while the right engine was operating at MCP. Zero sideslip was maintained for all tests. The climb drag polar equations for the C-12A aircraft are presented in table 4. Test results are presented in figures 21 through 29, appendix G.

Table 4. Climb Drag Polar Coefficients.[1]

Configuration	Number of Engines Operating	C_{D_o}	$\frac{\Delta C_D}{\Delta C_L^2}$	A	B	C
CR	Zero	0.03311	0.04237	Zero	Zero	Zero
	1	0.03311	0.04237	0.9139	.008197	-0.003456
	2	0.03311	0.04237	Zero	0.04818	-0.003418

[1]General drag equation: $C_D = C_{D_o} + \frac{\Delta C_D}{\Delta C_L^2} C_L^2 + AT_C'^2 + BT_C' + C$

Where:

C_D = Coefficient of drag

C_{D_O} = Minimum coefficient of drag of the propeller feathered drag polar

$\frac{\Delta C_D}{\Delta C_L{}^2}$ = Slope of drag polar

C_L = Coefficient of lift

$T_C{}'$ = Coefficient of thrust

A, B, C = Constants

13. At a gross weight of 12,000 pounds, the aircraft had a positive dual-engine rate of climb of 2400 feet per minute (ft/min) at the recommended best-rate-of-climb airspeed of 126 knots calibrated airspeed (KCAS) in the CR configuration at sea level on a standard day. The dual-engine cruise ceiling (300 ft/min rate of climb) was 28,600 feet, which fell short of the guaranteed cruise ceiling of 29,200 feet by 600 feet (2 percent).

14. At a gross weight of 12,000 pounds in the CR configuration, the aircraft had a positive single-engine rate of climb of 730 ft/min at the recommended best single-engine rate-of-climb airspeed of 117 KCAS at sea level on a standard day (15°C). The single-engine service ceiling was 16,580 feet, which was less than the guaranteed service ceiling of 17,600 feet by 6 percent. The single-engine performance capability of the C-12A under heavy gross weight and high temperature conditions (on a hot day) was 600 ft/min, which met the guarantee. The single-engine climb gradient (clean configuration) at 5000 feet on a standard day was 4.95 percent, which exceeded the guaranteed gradient of 3.45 percent.

Continuous Climb:

15. Dual and single-engine continuous climb performance was evaluated at the conditions shown in table 2. All continuous climbs were conducted using maximum continuous torque until reaching the engines' critical altitude and maximum continuous interstage turbine temperature throughout the remainder of the climbs. Single-engine continuous climbs were performed with the left engine shut down, the propeller feathered, and zero sideslip. The best-rate-of-climb airspeed schedule recommended by the contractor was used. The continuous climb test results verified the sawtooth climb test results and confirmed that the C-12A's single-engine service ceiling was 16,600 feet, which failed to meet the single-engine service ceiling guarantee of 17,600 feet by 1000 feet (6 percent).

Level Flight Performance

16. Level flight performance was evaluated at the conditions shown in table 2 to determine V_H, V_{CR}, range, and endurance capabilities. The zero thrust glide test method was used to obtain the base-line drag polar for the aircraft. The aircraft was stabilized and trimmed at incremental airspeeds in a descent with both engines inoperative and the propellers feathered. The constant pressure altitude technique was used for the determination of single-engine (propeller feathered) and dual-engine power required as a function of airspeed. The aircraft was stabilized and trimmed at incremental airspeeds from V_H to $1.1V_S$. Performance at conditions not specifically tested was calculated from the drag polar and power-available data, which included installation and accessories losses. The results of these tests are presented in figures 30 through 37, appendix G. Maximum airspeed, aircraft specific range, and recommended endurance in level flight for the CR configuration are summarized in figures 38 through 40. The level flight drag polar equations for the C-12A aircraft are presented in table 5.

Table 5. Level Flight Drag Polar Coefficients.[1]

Configuration	Number of Engines Operating	C_{D_o}	$\frac{\Delta C_D}{\Delta {C_L}^2}$	A	B	C
CR	Zero	0.03311	0.04237	Zero	Zero	Zero
	1	0.03311	0.04237	5.127	-0.4670	0.008927
	2	0.03311	0.04237	Zero	0.04109	-0.00452

[1]General drag equation: $C_D = C_{D_o} + \frac{\Delta C_D}{\Delta {C_L}^2} {C_L}^2 + A{T_C'}^2 + BT_C' + C$

17. Test results indicate that at a gross weight of 12,000 pounds at 30,000 feet, standard day, the maximum dual-engine airspeed in level flight using maximum cruise power available was 215 knots true airspeed (KTAS), which was less than the guarantee of 221 KTAS by 3 percent. The maximum level flight airspeed attainable at the maximum gross weight of 12,500 pounds was 225 KTAS at 14,000 feet. The maximum range airspeed was 215 KTAS and the endurance airspeed was 168 KTAS at 30,000 feet, standard day. Test results shown in table 6

indicate that the range guarantee (1045 nautical miles) was met. Tests with the ice vanes extended showed that a 7-percent increase in fuel flow was experienced. Further degradation of range performance should be anticipated for flights into icing conditions requiring the use of additional anti-icing devices or equipment.

Table 6. Range Performance.

Segment	Average True Airspeed (kt)	Fuel Used[1] (lb)	Altitude (ft)	Distance (naut mi)
Start,[2] taxi, accelerate to climb airspeed	Zero	86	Zero	Zero
Cruise climb to cruise altitude	185	360	Sea level to 30,000	118
Maximum cruise airspeed	221	1608	30,000	925
Fuel reserve[3]	153	215	30,000	Zero
Total		2269		1043

[1]Based on 6.7 pounds per gallon.
[2]Based on MCP for 5 minutes at sea level.
[3]Based on maximum endurance at 30,000 feet for 45 minutes.

18. At 12,500 pounds the maximum single-engine airspeed in level flight (left engine shut down, propeller feathered, and wings level), using single-engine cruise power available on the right engine at 9000 feet, was 180 KTAS on a standard day. The recommended single-engine airspeeds for maximum range and endurance at 12,000 pounds, sea level, standard day, were 162 and 109 KTAS, respectively. The results of these tests are presented in figures 41 through 43, appendix G. Test results for maximum airspeed, specific range, and recommended endurance in the single-engine CR configuration are summarized in figures 44 through 46.

Stall Performance

19. Dual and single-engine stall performance was evaluated at the conditions presented in table 2. The stall tests were initiated from the specified trim conditions by decelerating at approximately 1 knot per second until the airplane stalled. Stall was defined as moderate-to-heavy buffet accompanied by a high sink rate. Test results are presented in table 7 and two typical dual-engine stall time histories are presented in figures 47 and 48, appendix G.

Table 7. Representative Low-Altitude Stall Test Conditions and Performance.[1]

Configuration[2]	Angle of Bank (deg)	Power[3] (%)	Gross Weight (lb)	Calibrated Airspeed (kt)		
				Horn	Buffet	Stall
CR	Zero	Zero	12,770	110	106	105
CR	Zero	60	12,710	104	100	100
CR	Zero	100	12,690	100	98	96
CR	30 right	Zero	12,670	116	110	110
CR	30 left	Zero	12,640	112	109	109
CR	30 left	60	12,630	111	103	102
CR	45 left	Zero	12,600	123	117	116
TO[4]	Zero	Zero	12,680	107	105	104
TO[4]	Zero	60	12,640	101	96	95
TO[4]	Zero	100	12,620	99	88	88
PA	Zero	Zero	12,610	99	94	94
PA	Zero	60	12,570	88	84	84
PA	Zero	100	12,550	86	85	84
PA	30 left	Zero	12,660	106	100	98
PA	30 right	Zero	12,620	105	99	98
PA	30 right	60	12,570	94	87	86
PA	45 right	Zero	12,540	115	102	102
L	Zero	Zero	12,530	89	85	85
WO	Zero	60	12,490	81	74	73
WO	Zero	100	12,470	77	73	73
L	30 right	Zero	12,590	95	88	88
L	30 left	Zero	12,480	98	87	87
L	30 right	60	12,440	83	78	76
L	45 right	Zero	12,390	110	95	91

[1]Average density altitude: 10,630 feet. Ambient air temperature +4.6°C. Airplane cg: FS 196.5.
[2]All power-off stalls were repeated in each configuration with yaw damper engaged - test results were the same as those presented in the table for each configuration.
[3]Single-engine stall performance tests were conducted in each configuration, power OFF and ON, with no significant change in test results from those presented in the table for each configuration.
[4]Takeoff flap setting: Zero.

20. Complete stall performance tests were conducted at 10,000 feet density altitude (H_D) and spot-checked throughout the operational altitude envelope of the C-12A. The variation of the power-off stall airspeed with gross weight for unaccelerated and accelerated stalls for the C-12A was essentially as presented in the operator's manual.

21. The stall airspeed variation with power at 12,000 pounds gross weight and 10,000 feet altitude is presented in figure A. In the CR configuration, the stall airspeed only varied 5 to 6 knots from power-off to full power.

22. Another effect of the increase in stall airspeed with altitude was noted during maneuvers at 30,000 feet. In spite of the fact that the power-on stall airspeed was lower than the power-off airspeed, the margin between full power operation and stall was narrow at high altitude, thereby limiting the capability of the aircraft to perform normal maneuvers. At 30,000 feet the airspeed margin between maximum cruise airspeed (134 KCAS) and power-on stall (116 KCAS) was only 18 knots, and was narrowed to 10 knots in a 30-degree banked turn. In a worse case, the V_{CR} climb at 120 KCAS had a margin of only 4 knots, and a 15-degree banked climbing turn was all that was required to enter an accelerated power-on stall. A discussion of the effects on stall performance of altitude, power, and maneuvers is not included in the operator's manual and it should be amended to include such.

23. The CR and TO configuration power-off stall airspeeds were within 1 knot of each other. However, with power the difference in stall airspeed for the two configurations became significant (8 knots at full power). This increase in stall airspeed with the landing gear retracted (94 KCAS in CR versus 86 KCAS in TO at 12,000 pounds gross weight) has particular signficance during short field takeoffs and should be included in an added discussion of short field techniques in the operator's manual.

24. Within the scope of these tests, the stall performance of the C-12A is satisfactory throughout the certified operational flight envelope of the aircraft.

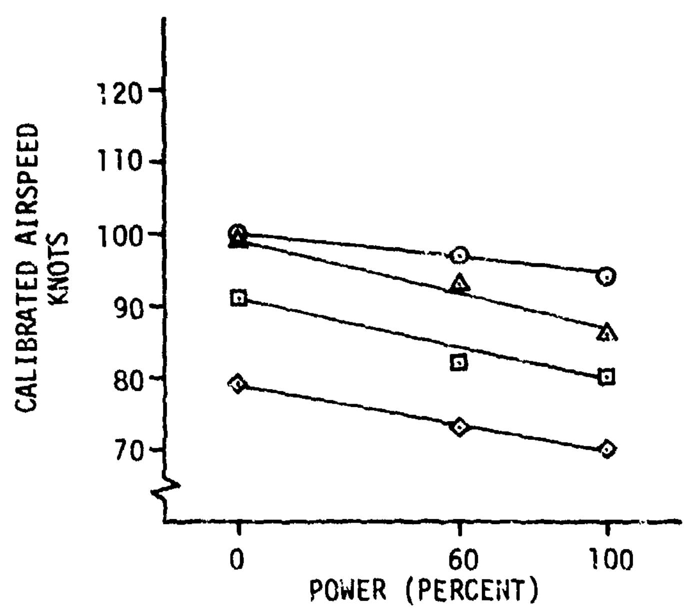
HANDLING QUALITIES

General

25. The handling qualities of the C-12A were evaluated under a variety of operating conditions, with emphasis on operation at the maximum gross weight and at the expected most critical condition of aft cg loading with the autopilot OFF. The test airplane handling qualities were compared to FAR Part 23 and MIL-F-8785B(ASG). Two enhancing characteristics, the rudder boost system and the location of the landing light switches; two deficiencies, the wheel lockup tendency during landing and the lack of artificial stall warning above 20,000 feet; and four handling qualities shortcomings were noted during the test. The shortcomings were related to the poor trimmability of the airplane, yaw damper

FIGURE A.
STALL AIRSPEED
VARIATION WITH POWER

O CR CONFIGURATION
Δ TO CONFIGURATION
□ PA CONFIGURATION
◇ WO CONFIGURATION
12000 LB GROSS WEIGHT
GS = FS 196.5



operation during high-speed flight in turbulence, lighting systems, aircraft systems operation, and equipment failures during the tests. Of the shortcomings, 14 were associated with reliability and maintainability.

Control System Characteristics

26. Control system characteristics were evaluated in flight at the conditions shown in tables 2 and 3. Control forces were measured on the pilot control wheel and rudder pedals. The cockpit controls versus control surface positions obtained during ground calibration and rigging check are presented in figure 49, appendix G. Control system positions in trimmed forward flight are presented in figures 50 through 57. There was no detectable lag in aircraft response to either small or large control inputs about any control axis. Elevator and aileron force and displacement sensitivities were compatible and intentional inputs to one control axis did not cause inadvertent inputs to another axis. Control harmony was good and there was no tendency for the pilot to induce undesirable motion. However, moderate departures from trim conditions (6 knots) did occur with the controls free, due to the friction band, weak static longitudinal stability, and light phugoid damping, all encountered at trim conditions throughout the airspeed envelope. The poor trimmability was objectionable and constitutes a shortcoming (HQRS 4). However, with the autopilot engaged the trimmability is satisfactory.

Static Longitudinal Stability

27. Static longitudinal stability characteristics were evaluated at the conditions shown in table 3. The aircraft was trimmed in steady-heading, ball-centered level flight at the desired trim airspeed, then stabilized at incremental airspeeds greater than and less than trim airspeeds. Test results are presented in figures 58 through 69, appendix G.

28. The stick-free static longitudinal stability, as indicated by the variation in elevator control force with airspeed, was positive for airspeeds both above and below trim in all configurations tested, indicating stable static longitudinal stability. In all configurations at aft cg locations a lightening of the elevator control forces for airspeeds below trim was noted. This lightening of control force was objectionable for maintaining precise airspeed control (HQRS 4) and is a shortcoming.

29. The stick-fixed static longitudinal stability at a forward cg, as indicated by the variation in elevator control position with airspeed, was positive, although extremely shallow, for airspeeds above and below trim. The elevator control position gradient was essentially neutral at aft cg locations for all configurations tested. The neutral elevator control position gradient is undesirable when coupled with the small breakout forces and resulting high sensitivity of the elevator control. The static longitudinal stability characteristics met the requirements of paragraphs 23.173 and 23.175 of FAR Part 23 but failed to meet the requirements of paragraph 3.2.1.1 of MIL-F-8785B(ASG), in that the elevator

control position gradient at aft cg locations is essentially neutral. Although the stick-fixed requirements of MIL-F-8785B(ASG) were not met, the static longitudinal stability characteristics of the C-12A aircraft are acceptable.

Static Lateral-Directional Stability

30. The static lateral-directional stability characteristics of the C-12A airplane were evaluated at the conditions shown in table 3. The aircraft was initially trimmed for zero sideslip at the desired airspeed. The aircraft was then stabilized at incremental sideslip angles left and right at constant airspeed and heading. Maximum sideslip attained was limited in the PA configuration by a divergent Dutch roll and in the CR configuration by rudder control forces. Test results are presented in figures 70 through 74, appendix G.

31. Static directional stability, as indicated by the variation of sideslip angle with rudder pedal deflection, was positive for sideslip angles between 5 degrees, left and right, from trim. A lightening of rudder pedal force with increasing rudder pedal deflection occurred at sideslip angles well outside this range at all airspeeds; however, the rudder pedal force never reduced to zero. In the CR configuration at airspeeds above 148 KCAS, the maximum sideslip attainable was limited by rudder pedal forces in excess of 150 pounds. In the PA configuration at low altitude and in the CR configuration at 25,000 feet H_D, maximum sideslip was limited by a divergent Dutch roll which coupled with pitch oscillation. This oscillation had a damping ratio of δ_d = -0.01 and an undamped natural frequency of ω_{n_d} = 1.48 radians/second (0.23 Hz). This phenomenon occurred at approximately 7-1/2 degrees angle of bank. A representative time history of this oscillation is presented in figure 75, appendix G. Recovery from this oscillation was immediately effected by decreasing rudder deflection. The oscillation presented no problem in the CR configuration, in that it occurred at a point well beyond normal maneuvering limits; however, final runway alignment during approaches with crosswind components near the 25-knot limit required sideslips which approached the boundary of this oscillation. During these approaches the handling qualities of the airplane were improved by making the approach without flaps and maintaining an approach airspeed of 120 KIAS until just prior to touchdown. Because of the possibility of encountering this oscillation during crosswind approaches, the following CAUTION should be incorporated in the operator's manual:

CAUTION

Approaches with a crosswind component in excess of 20 knots should be made with flaps up. An approach airspeed of 120 KIAS should be maintained untill just prior to touchdown.

32. Further evaluation of the static directional stability during aileron-only turns revealed that during normal maneuvering, the aircraft could be flown as a two-control (aileron and elevator) airplane. Above 15,000 feet H_D, a minor Dutch-roll oscillation ensued following abrupt aileron movements. With the yaw

damper disengaged, the Dutch roll did not present a control problem and, depending on altitude, damped out after several oscillations. With the yaw damper engaged, the oscillation damped in two to three cycles. Maneuvers simulating low-altitude realignment with the runway after breaking out of the clouds on a nonprecision instrument approach were performed. During these maneuvers, minimal rudder coordination was required with the yaw damper OFF (HQRS 3) and with the yaw damper ON rudder was not required to minimize adverse yaw and the ensuing Dutch roll (HQRS 2).

33. The variation of sideslip angle with rudder pedal deflection was essentially linear for all sideslip angles tested. Within the scope of this test, the static directional stability met the requirements of paragraphs 23.177(a)(1) and (3) of FAR Part 23. In no case did the rudder pedal control force approach zero. The static directional stability did meet the requirements of MIL-F-8785B(ASG) and was acceptable.

34. Dihedral effect, as indicated by the variation of aileron control displacement with sideslip angle, was positive and essentially linear. Some nose-down pitch coupling was present, as indicated by the requirement for increasing aft elevator control displacement and force with increasing sideslip angles in both directions; however, aileron and elevator control force harmony was excellent and the forces were acceptable. Further evaluation using rudder-only turns confirmed strong dihedral effect, in that bank angle was easily controlled by small rudder displacements. Small heading changes were easily accomplished during approaches using rudders only (HQRS 2).

35. The side-force characteristics, as indicated by the variation of bank angle with sideslip angle, were positive and essentially linear for all configurations tested. The side-force characteristics provided the pilot with good cues of out-of-trim flight conditions. Within the scope of this test, the static lateral-directional stability characteristics of the C-12A airplane are satisfactory.

Dynamic Longitudinal Stability

36. The dynamic longitudinal stability characteristics were evaluated at the conditions shown in table 3. The long-term (phugoid) dynamic characteristics were evaluated by reducing or increasing airspeed approximately 15 knots with the elevator control and then releasing the control, allowing it to seek the trim position (stick-free). The longitudinal short-term characteristics were evaluated by rapidly displacing the elevator control in a 1-inch doublet, releasing the control at the trim position. Time histories of representative dynamic response characteristics are presented in figures 76 and 77, appendix G.

37. The phugoid response was oscillatory, easily excited, and lightly damped for all configurations tested except PA. The periods varied from approximately 55 seconds in CR to 42 seconds in approach. In the PA configuration, releases from airspeeds below trim failed to excite the long-term dynamic aircraft mode. The extremely shallow elevator position gradient with respect to airspeed and lack of absolute centering prevented the elevator control from returning to the trim

position. Although there was no procurement specification requirement for phugoid stability, the long-term characteristics failed to meet the requirements of paragraph 3.2.1.2 of MIL-F-8785B(ASG), in that the damping ratio of the phugoid oscillation was less than 0.04. This weak damping contributed to the poor trimmability but the long-term longitudinal dynamic characteristics are acceptable.

38. The longitudinal short-term characteristics were oscillatory, of low natural frequency, and heavily damped. The short-term characteristics met the requirements of paragraph 23.181 of FAR Part 23 and of MIL-F-8785B(ASG). For the conditions investigated, the short-term longitudinal dynamic characteristics are acceptable.

Dynamic Lateral-Directional Stability

Dutch-Roll Characteristics:

39. The dynamic lateral-directional stability characteristics of the C-12A were evaluated at the conditions presented in table 3. The Dutch-roll characteristics were evaluated by exciting the aircraft with rudder doublets and releases from steady-heading sideslips. Tests were conducted with the yaw damper ON and OFF and with controls fixed and free. Time histories of representative dynamic lateral-directional airplane responses are presented in figures 78 through 84, appendix G. Test results are summarized in table 8.

40. The Dutch roll was lightly damped and easily excited. In smooth air the Dutch roll tended to damp out in four to five oscillations with the yaw damper OFF and one and one-half oscillations with the yaw damper ON. The aircraft lateral-directional response and controllability characteristics were poor in the presence of turbulence with the yaw damper OFF. Considerable pilot compensation was required to damp the Dutch roll using primary flight controls (HQRS 5). Satisfactory performance was obtained with the yaw damper and/or autopilot engaged in all flight regimes except descents in turbulence at airspeeds greater than 200 KIAS. With the yaw damper engaged at airspeeds greater than 200 KIAS, the damping of the Dutch roll decreased to a value below the natural damping of the unaugmented airplane and the yaw damper drove the Dutch roll divergent. Each time this was encountered, the aircraft recovered as soon as the yaw damper was disengaged. The tendency of the yaw damper to drive the Dutch roll divergent in the presence of turbulence at airspeeds in excess of 200 KIAS is a shortcoming. Until the problem of the yaw damper is corrected, the following CAUTION should be incorporated in the operator's manual:

CAUTION

> Disengage the yaw damper during descent in turbulence at airspeeds in excess of 200 KIAS if at any time the aircraft lateral or directional oscillations begin increasing in amplitude. Disengagement of the yaw damper will allow the airplane to recover itself.

Table 8. Dutch-Roll Characteristics.

Calibrated Trim Airspeed (kt)	Configuration	Average Density Altitude (ft)	Damping Ratio (ζ)	Undamped Natural Frequency ω_{nd} (rad/sec)	Period (τ sec)	Roll/Yaw Ratio (ϕ/β)
Yaw Damper OFF						
120	PA fixed	10,290	0.09	1.50	4.20	1.05
150	CR fixed	9880	0.09	1.90	3.30	1.00
181	CR fixed	10,230	0.09	2.20	2.80	1.13
180	CR fixed	10,230	0.09	2.30	2.70	1.13
171	CR fixed	25,080	0.05	2.30	2.75	1.38
169	CR fixed	25,220	0.06	2.25	2.80	1.33
Yaw Damper ON						
121	PA fixed	10,150	0.3	1.50	4.30	0.87
119	PA free	10,130	0.3	1.75	3.80	0.91
147	CR fixed	9630	0.22	2.10	3.00	1.08
151	CR free	9560	0.25	2.32	2.80	1.00
179	CR fixed	10,180	0.09	2.55	2.50	1.25
180	CR free	9950	0.10	2.53	2.50	1.18
270	CR free	10,230	0.02	3.94	1.63	1.30, 1.60
170	CR fixed	25,340	0.12	2.30	2.75	1.33
170	CR fixed	25,400	0.12	2.30	2.70	1.31

41. The Dutch roll was lightly damped with the yaw damper OFF and failed to meet the requirements of FAR Part 23, paragraph 23.177a(4); however, with the yaw damper engaged, damping was satisfactory at all airspeeds except in high-speed descent, as mentioned above.

Spiral Stability Characteristics:

42. The spiral stability characteristics of the C-12A were evaluated by establishing trimmed level flight conditions and then stabilizing in a 10-degree bank angle, using rudders only. Once the bank angle was established, the rudder pedal was slowly returned to trim and the resulting tendency of the aircraft to increase or decrease bank angle was noted. The tests were conducted at the conditions shown in table 3 and test results are presented in table 9.

43. Spiral stability was neutral to slightly divergent in almost all instances, regardless of configuration, yaw damper operation, or altitude. The slight divergence of the spiral mode was confirmed by the requirement for aileron out of the turn in constant angle of bank, aileron-only turns, and top rudder in constant bank angle rudder-only turns. Within the scope of this test, the spiral stability characteristics met the requirements of MIL-F-8785B(ASG) and are satisfactory.

Maneuvering Stability

44. Maneuvering stability characteristics were evaluated at the conditions shown in table 3. The variation of elevator control force and control position with normal acceleration was determined by trimming the aircraft in coordinated level flight at the desired trim airspeed and then stabilizing at incremental bank angles in steady turns, both left and right. Airspeed and power were held constant and the aircraft was allowed to descend during the maneuver. Data were obtained at each stabilized bank angle. Symmetrical pull-up maneuvers were used to obtain load factors in excess of 2.0 and symmetrical pushover maneuvers to obtain data below +1g. The load factor was varied incrementally to the maximum allowable during these maneuvers. The test results of the maneuvering stability evaluation are presented in figures 85 through 89, appendix G.

45. The stick-free maneuvering stability, as indicated by the variation of elevator control force with normal acceleration, was positive (increased aft elevator control force with increased load factor) and was essentially linear for all conditions tested. The elevator control force gradient (stick force per g) was approximately 30 pounds per g. Buffeting was encountered while attempting to achieve load factors in excess of 2.0 at 140 KIAS. The maneuvering control force gradients were sufficiently high to prevent control inputs which could produce undesirably abrupt aircraft response or tendencies toward pilot-induced oscillations.

46. The stick-fixed maneuvering stability, as indicated by the variation of elevator control position with normal acceleration, was slightly positive (increased aft elevator control motion with increased load factor) but shallow, and essentially linear. The control position gradient was approximately 0.4 inch per g. This

Table 9. Spiral Stability Characteristics.

Calibrated Trim Airspeed (kt)	Configuration	Density Altitude (ft)	Direction of Bank	Time (sec)
Yaw Damper OFF				
123	PA	10,610	Left	[1]52
118	PA	9880	Right	[2]NA
150	CR	9490	Left	NA
151	CR	9430	Right	NA
173	CR	25,100	Left	[1]72
171	CR	25,000	Right	NA
181	CR	10,010	Left	[3]25
180	CR	10,050	Right	NA
Yaw Damper ON				
125	PA	10,360	Right	[1]57
123	PA	10,570	Left	[1]82
172	CR	25,200	Right	[3]48
170	CR	25,310	Left	NA

[1]Time to double bank angle.
[2]Not applicable.
[3]Time to one-half bank angle.

shallow control position gradient was objectionable when maintaining steady turn conditions (HQRS 6). Compounding this problem were the easily excited long period, light breakout forces, and the divergent spiral mode which occurred at bank angles greater than 10 degrees. The maneuvering stability characteristics of the C-12A met the requirements of paragraph 3.2.2 of MIL-F-8785B(ASG), except paragraph 3.2.2.2.2, in that the shallow elevator control position gradient in maneuvering flight is a shortcoming.

Roll Performance

47. Roll performance of the C-12A was evaluated at the conditions presented in table 3. These tests were initiated from a trimmed unaccelerated flight condition by applying both one-half deflection and full deflection aileron control inputs (in 0.2 second) without changing either elevator or rudder pedal control position. Test results are presented as representative time histories of airplane response with one-half and full deflection aileron inputs in figures 90 through 94, appendix G, and summarized for the full deflection rolls in table 10.

48. In full deflection rolls with rudder fixed and free, the maximum adverse yaw decreased as airspeed was increased. The maximum adverse yaw generated was 7 degrees for a full deflection roll in the PA configuration at 120 KCAS. In no instance, with yaw damper ON or OFF and controls fixed or free, was the adverse yaw objectionable. The Dutch roll was very slightly excited during these maneuvers with the yaw damper OFF and was heavily damped with the yaw damper ON. The aircraft was very responsive in roll for this category aircraft without any apparent cross-coupling and the roll and pitch control harmony was excellent. The roll performance characteristics met the requirements of MIL-F-8785B(ASG). Within the scope of these tests, roll performance characteristics are satisfactory.

Stall Characteristics

General:

49. Dual and single-engine stall characteristics of the C-12A airplane were evaluated at the conditions listed in table 2. Stalls were initiated from the specified trim conditions by decelerating at a rate of approximately 1 knot per second until the airplane stalled. Stall warning, stall, and stall recovery characteristics were evaluated. Dual-engine stalls were evaluated with power OFF, partial power (60 percent torque per engine), and high power (100 percent torque at 10,000 feet and maximum attainable torque at higher altitudes). Single-engine stalls were evaluated with the critical engine (left engine) shut off, propeller feathered, and MCP on the remaining engine.

Stall Warning:

50. Initial stall warning for all stalls below 15,000 feet was provided by the stall warning horn. A lift transducer mounted in the leading edge of the left wing transmits a signal to a lift computer incorporated in the stall warning system to

Table 10. Roll Performance.

Configuration	Calibrated Trim Airspeed (kt)	Aileron Control Displacement	Adverse Yaw (deg)	Maximum Roll Rate (deg/sec)	Roll Mode Time Constant	Time to 60 Degrees (sec)	Average Aileron Control Force (lb)	Nondimensional Roll Rate Ratio[1]
PA fixed[2]	121	1/2 left	5	22.0	0.35	3.30	30	0.042
PA fixed	118	1/2 right	4	20.0	0.35	3.60	30	0.039
PA free[3]	120	1/2 right	4.5	20.0	0.35	3.60	22	0.038
PA free	120	1/2 left	6.0	20.0	0.35	3.30	27	0.038
PA free[4]	120	1/2 left	7.0	20.0	0.35	3.30	27	0.039
PA free[4]	118	1/2 right	5.5	20.0	0.35	3.30	30	0.039
PA fixed	124	Full left	7.0	42.5	0.35	1.85	60	0.079
PA fixed	126	Full right	6.0	47.0	0.37	1.85	65	0.086
PA free[4]	124	Full left	6.5	42.5	0.35	1.85	60	0.079
PA free[4]	125	Full right	7.0	48.0	0.35	1.80	70	0.088
CR fixed	157	1/2 right	3.0	25.0	0.30	3.10	35	0.036
CR fixed	154	1/2 left	3.5	25.0	0.30	2.70	40	0.037
CR fixed	156	Full left	6.0	55.0	0.30	1.55	70	0.081
CR fixed	155	Full right	5.0	57.5	0.30	1.55	75	0.085
CR fixed	157	Full left	5.0	55.0	0.30	1.50	70	0.080
CR fixed	183	1/2 right	2.0	24.0	0.25	2.95	45	0.030
CR fixed	183	1/2 left	2.0	27.5	0.25	2.40	45	0.034
CR free	182	1/2 right	2.0	25.0	0.25	2.90	45	0.031
CR fixed	184	Full left	4.0	63.0	0.25	1.35	70	0.078
CR fixed	182	Full right	3.5	63.0	0.25	1.35	75	0.079

[1] $\frac{Pb}{2V_T}$

[2] Controls fixed following input.

[3] Controls free following input.

[4] Yaw damper ON.

provide a programmed stall warning margin above stall. Below 10,000 feet, the stall warning horn provided excellent warning at approximately 4 to 10 knots above stall, depending on power and configuration. At 20,000 feet, the stall warning coincided with buffet onset and above that altitude airframe buffet occurred first, followed by stall warning horn activation and stall. The angle of attack for stall warning horn activation was invariant with power, varied with configuration (increased 2 degrees when flaps were lowered), and varied with altitude, as shown in figure B.

51. Artificial stall warning above 15,000 feet is necessary because the natural cues to the approach of stall (pitch attitude and control effectiveness) were insufficient to provide adequate warning. Airplane pitch attitudes in a full power stall (18 to 19 degrees, nose up) were not significantly different from those achieved during normal maneuvers at best-rate-of-climb and cruise climb airspeeds above 25,000 feet (16 to 17 degrees, nose up) or transient pitch attitudes during short field takeoffs and obstacle clearance climbs (20 to 30 degrees, nose up). Control effectiveness was excellent until buffet onset, with only a slight decrease in control effectiveness once airframe buffet had commenced, thus providing very little cue to the approach of stalls. The insufficient stall warning margin between 15,000 and 20,000 feet is a shortcoming and the lack of suitable stall warning above 20,000 feet is a deficiency.

52. The most consistent indication of stall was the onset of moderate buffet coupled with a slight decrease in the effectiveness of all controls. Buffet onset was distinct and unmistakable. Buffet was followed very closely by stall (1 to 2 knots) and, dependent on power and configuration, was frequently coincident with stall. For this reason buffet was not suitable as the sole means of stall warning.

53. The stall warning characteristics of the C-12A failed to meet the following requirements:

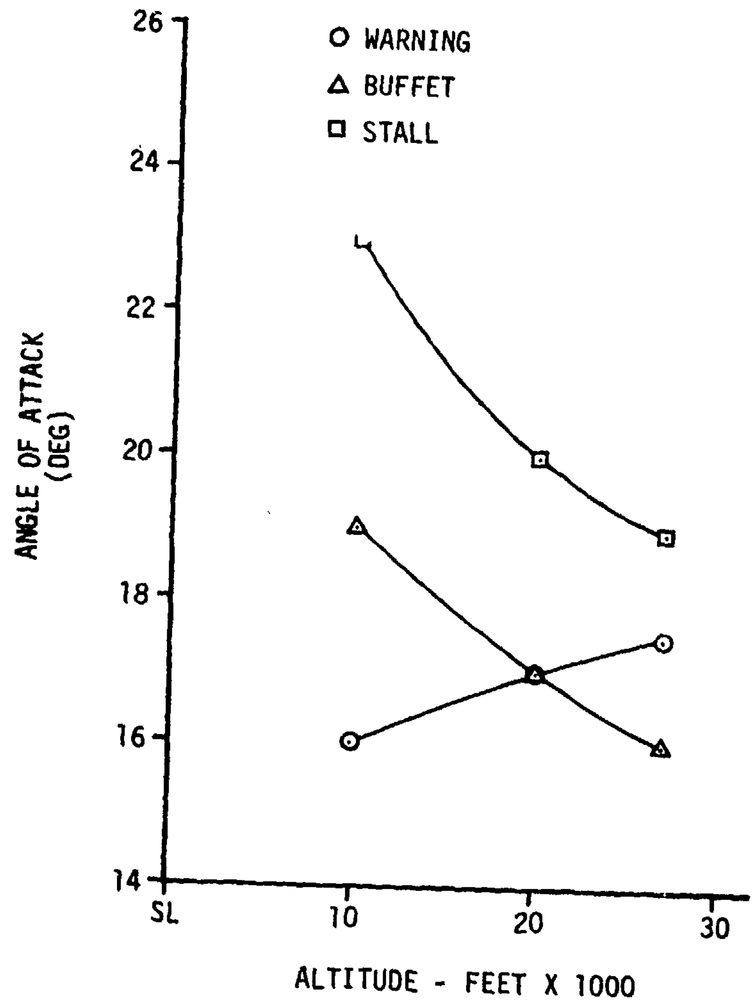
a. FAR Part 23:

(1) Paragraph 23.207b - The stall warning did not occur at an airspeed at least equal to V_S plus 5 knots. The airplane failed to meet the requirements of this paragraph in all configurations above 10,000 feet and below 10,000 feet by 1 knot in CR configuration, power ON; 2 knots in TO configuration, power OFF; 3 knots in PA configuration, power ON; 1 knot in L configuration; and 1 knot in WO configuration.

(2) Paragraph 23.207c - A clear and distinct warning did not always exist. Above 20,000 feet the stall warning horn sometimes activated poststall.

b. MIL-F-8785B(ASG), paragraph 3.4.2.1.1.1 - The minimum airspeed for warning onset was less than V_S plus 5 knots, as listed in subparagraph a(1) above.

FIGURE B.
STALL ANGLE OF ATTACK
VARIATION WITH ALTITUDE
CR CONFIGURATION, POWER OFF



Stall:

Unaccelerated Stalls

54. Stall in the C-12A was not as distinct as it is in most other light airplanes. The stall was characterized by several subtle occurrences: (1) a slight increase in buffet intensity; (2) a slow yaw divergence; (3) a slight pitch oscillation; (4) a tendency toward wing rock which required large, frequent aileron inputs to control; (5) a significant increase in rate of sink, as high as 8500 ft/min in deep stall; (6) a slow decrease in elevator control effectiveness, allowing full elevator control to be reached in deep stall; (7) a slow increase in angle of attack to approximately 60 degrees in deep stall; (8) extremely erratic ship's system airspeed indications; and (9) an initially high pitch attitude at stall entry (16 to 17 degrees), settling down to a level attitude as deep stall was attained. At stall the airplane almost always developed large sideslip angles, as high as ±20 degrees. These large sideslips were difficult to control even with large rudder deflections. Even with these large sideslip angles the airplane exhibited little tendency to depart. The only departure during unaccelerated stall tests occurred at 20,000 feet, power ON, in the PA configuration where pro spin controls, right rudder and left aileron, were used in an effort to control sideslip and roll. Even though this departure is potentially dangerous, it occurred well beyond the limits of normal flight maneuvers and should not be encountered by the operational pilot.

55. Normal stall was accompanied by rates of sink up to 3000 ft/min, power OFF, and zero to 840 ft/min at full power. Deep stall was easily recognized because it was a very stable flight regime marked by a level pitch attitude, full aft elevator control, and a characteristic rate of sink up to 8500 ft/min regardless of the power applied. In deep stall full aft elevator could be held with only fingertip pressure (approximately 5 pounds), turns were possible using ailerons, and the airplane exhibited near neutral apparent directional stability. Because of the directional control problems, the airplane was intentionally yawed to ±20 degrees of sideslip to determine if any departure tendencies existed. During these sideslip excursions the airplane held whatever sideslip was induced, with no tendency to either depart or return to zero sideslip. These deep stall characteristics were the same regardless of cg location, the only difference being that deep stall was more easily attained at a forward cg. In general, the C-12A unaccelerated stalls were free from any adverse departure, poststall gyration, or spin tendencies and were satisfactory.

56. Altitude generally degraded the stall characteristics described, making roll and yaw slightly harder to control. Yaw damper operation generally induced right sideslip because the yaw damper maintained the trim rudder position where it had been engaged. Also, the yaw damper significantly hampered the pilot's efforts to control sideslip and as much as 90 to 120 pounds of rudder force was required to move the rudder to correct sideslip resulting from airspeed changes (HQRS 5). The yaw damper also hampered stall recovery in a similar manner. Aircraft control was enhanced once the yaw damper was disengaged. The following NOTE should be included in the operator's manual:

NOTE

During practice stalls with the yaw damper engaged the pilot may experience pedal forces up to 120 pounds to maintain balanced flight. Stall recovery can be significantly improved by disengaging the yaw damper.

Accelerated Stalls

57. Accelerated stalls were evaluated with power ON and OFF in the CR, TO, PA, WO, and L configurations in 30, 45, and 60-degree banked turns left and right. At stall, the airplane exhibited the same characteristics as in the unaccelerated stall tests, with the exception that the elevator forces were high (10 to 15 pounds in a 30-degree banked turn and 25 to 30 pounds in a 45-degree banked turn) and there was a tendency for the airplane to roll out of the stall, pro recovery, requiring the pilot to hold the airplane in the turn. The roll tendency increased with angle of bank. This inherent roll tendency initiated recovery (HQRS 3). Stalls at angles of bank greater than 30 degrees were difficult to achieve because of the high elevator forces. In all accelerated stalls below 15,000 feet H_D adequate warning was present in the form of the artificial warning horn and the airframe buffet. Below 15,000 feet, the high elevator forces are sufficient to prevent inadvertent excursions into stall and the normal maneuvers in the C-12A rarely require angles of bank over 30 degrees. Accelerated stalls were inadvertently encountered at altitudes above 20,000 feet H_D. For higher altitudes, the following CAUTION should be included in the operator's manual:

CAUTION

Constant altitude turns above 20,000 feet and airspeed less than 130 KCAS may result in stall at very small bank angles.

Single-Engine Unaccelerated Stalls

58. Single-engine unaccelerated stalls with either engine out exhibited essentially the same characteristics as dual-engine stalls. Stall was always at or below static V_{MC} with full power applied. Full rudder deflection was insufficient to prevent sideslip, which resulted in all full-power single-engine stalls being characterized by a roll coupled with a nose-down pitch. Airplane response was most pronounced in single-engine stalls with the left engine inoperative. Power ON, the stall with the left engine shut off was characterized by a rapid left roll, which increased in rate when increased power was applied to the operating engine. Power-on single-engine stalls were the only stalls where the airplane was prone to departure. The following WARNING of the possible consequences of single-engine stalls should be included in the operator's manual.

WARNING

Single-engine stalls must be avoided. A high roll rate toward the dead engine will develop at stall. This roll rapidly progresses into a complete wing-over and will result in an altitude loss of 2000 feet or more prior to recovery.

Stall Recovery:

59. All normal dual-engine stalls were recovered by relaxing aft control force, returning the airplane to level flight attitude with the nose on or slightly below the horizon, and adding power to minimize altitude loss. Rapid recovery was hampered by a pronounced secondary stall tendency (recurrence of buffet), regardless of power or configuration. The secondary stall tendency was avoided by increasing airspeed at least 15 knots above stall before applying any aft elevator force to arrest the rate of sink. Altitude loss during normal stall recovery was generally on the order of 300 to 700 feet. The greatest loss of altitude occurred in the power-off stalls because of the engine acceleration time.

60. Recovery from dual-engine deep stalls generally resulted in an average loss of 800 feet, and was the only stall which required a 10 to 15-degree nose-down pitch attitude to break the stall before recovery could be effected. At high altitude, the altitude loss during recovery was increased significantly and almost doubled above 20,000 feet, with an average of 1000 feet of altitude lost in recovery.

61. Single-engine stall recovery was best achieved by slightly reducing power on the operating engine at onset of buffet, lowering the nose of the aircraft below the horizon, and accelerating rapidly to best single-engine rate-of-climb airspeed, then coordinating maximum controllable power to minimize altitude loss. Altitude loss during single-engine stall recovery was an average of 800 feet. A discussion of the normal, deep, and single-engine stall recovery techniques and associated altitude losses should be included in the operator's manual.

Single-Engine Characteristics

62. The single-engine handling qualities of the C-12A were evaluated at the conditions presented in table 3. With the left engine shut down and the propeller feathered, the airplane was decelerated at 1 knot per second, wings level, until a control limit or stall was reached. The airplane was then banked 5 degrees into the good engine, control deflection adjusted to maintain steady heading, and the deceleration commenced again until a control limit or stall was reached, establishing the static V_{MC} for the airplane. The minimum dynamic V_{MC} was next determined as the lowest possible airspeed at which the pilot could regain and maintain steady straight flight in any configuration following a sudden complete failure of the critical engine. All flight controls were used to effect recovery; however, power was not reduced on the operating engine, trim was not changed, and the propeller was not feathered. To allow for pilot reaction and recognition time, 2 seconds or a 20-degree angle of bank change, whichever came first, was allowed before

any corrective action was taken. For additional safety margin, the airspeed selected as the minimum dynamic V_{MC} was that airspeed where 90 percent control deflection, excessive control force, or a 20-degree heading change were required to regain and maintain control. Test results are presented in tables 11 and 12 for the static and dynamic cases, respectively.

Static Single-Engine Minimum Control Airspeed:

63. Static single-engine V_{MC} in wings-level flight was defined by loss of directional control as a result of achieving full rudder deflection. Static single-engine V_{MC} with the aircraft banked 5 degrees into the good engine was defined by airframe stall in the CR and TO configurations and full right aileron and rudder deflection in the WO configuration. Minimum airspeed for trim effectiveness (V_{min} trim) was 97 KCAS in the CR configuration, 95 KCAS in the TO configuration, both with the right rudder trim at its limit, and 102 KCAS in the WO configuration with the right aileron and rudder trim at their limit. In all configurations except WO, the aircraft stalled at the static V_{MC} with the characteristics described in paragraph 49. Airframe buffet provided an excellent cue to the pilot that single-engine V_{MC} had been achieved. With the rudder boost OFF, rudder control forces were high near static V_{MC}, often in excess of 100 pounds. With rudder boost ON, these forces were reduced, often to as little as 10 pounds, which is an enhancing characteristic. In all cases airplane control was easily maintained in all axes down to static V_{MC}.

Dynamic Single-Engine Minimum Control Airspeed:

64. Dynamic V_{MC} was established by the ability of the pilot to regain and maintain control of the airplane after a sudden failure of the critical engine. Transient rudder control forces at dynamic V_{MC} with the rudder boost and yaw damper OFF were as high as 180 pounds (125 pounds sustained). With the rudder boost only engaged, the transient forces were reduced 20 to 40 pounds and the sustained rudder pedal forces seldom exceeded 50 pounds. With rudder boost only engaged, directional control could easily be maintained following an unexpected engine failure. The rudder boost automatically made a 10 to 12-percent rudder deflection input within 1.5 to 2 seconds. This provided an excellent cue to the proper response and reduced pilot workload by reducing the normally high pedal forces during asymmetric power conditions. The rudder boost system is an enhancing feature which should be incorporated in future designs.

65. With the yaw damper engaged and rudder boost disengaged, rudder response was immediate following engine failure, and the damper deflected the rudder surface 20 percent in the proper direction. Although this input greatly improved pilot recognition and reaction time by providing an excellent cue to the pilot of the proper control inputs required to correct for the failed engine, the significant control force required to overcome the damper servo (200 pounds transient and 165 pounds sustained) hampered complete recovery by limiting the size of the final pilot input (HQRS 5). The rudder boost in combination with the yaw damper reduced the rudder control force to a manageable level and enhanced recovery

Table 11. Single-Engine Static Minimum-Control Airspeed.[1]

Configuration	Wings-Level			5-Degree Angle of Bank		
	Calibrated Airspeed (kt)	Limiting Factor	Peak Rudder Force (lb)	Calibrated Airspeed (kt)	Limiting Factor	Peak Rudder Force (lb)
CR	97	Rudder and stall	100	Note[3]		
CR RB[2]	96	Rudder and stall	10			
TO	97	Rudder	100	93	Rudder and stall	80
TO RB	96	Rudder	30	90	Stall	30
WO	91	Rudder	100	76	Aileron and rudder	80
WO RB	91	Rudder	15	75	Aileron and rudder	15

[1]Gross weight, 12,000 pounds; cg location, 196.4 inches.
[2]Rudder boost ON.
[3]No data available because the aircraft stalled wings-level.

Table 12. Single-Engine Dynamic Minimum-Control Airspeed.[1]

Configuration	Calibrated Airspeed (kt)	Rudder Surface Position (%)	Aileron Surface Position (%)	Peak Transient Rudder Force (lb)	Sustained Rudder Force (lb)	Maximum Roll Attitude (deg)	Maximum Transient Heading Change (deg)	Sustained Heading Change (deg)
CR	95	77	100	150	125	30	28	18
CR RB[2]	95	88	100	100	50	30	27	10
CR YD[3]	95	58	97	200+	165	24	20	20
CR RB YD	95	85	100	190	120	22	20	12
TO	95	69	100	180	120	26	25	20
TO RB	95	88	94	120	50	22	25	20
WO	93	88	100	160	100	28	22	18
WO RB	92	92	100	130	50	30	20	20

[1]Gross weight, 12,000 pounds; cg location, 196.4 inches.
[2]Rudder boost ON.
[3]Yaw damper engaged.

to the point where the pilot could satisfactorily fly the aircraft until he could disable the yaw damper (HQRS 4). Because of the high forces associated with working against the yaw damper, the following CAUTION should be added to the operator's manual:

CAUTION

> If a single-engine failure occurs during flight with the yaw damper engaged, high rudder pedal forces will be encountered (+200 pounds). Rapid disengagement of the yaw damper will enhance the pilot's ability to control the aircraft.

66. As shown by the test results, the single-engine V_{MC} airspeed (87 KIAS) presented in the operator's manual is unrealistic and unsafe and should be deleted and replaced by the information presented in table 11. Also, the red radial at 87 KIAS on the airspeed indicator in the airplane should be removed. The airspeeds presented in table 11 are the minimum for the maximum gross weight of 12,500 pounds and as such should provide a safety margin for lighter gross weights. Altitude loss from initial engine failure through recovery ranged from 200 to 400 feet for all configurations tested. Within the scope of this test, the single-engine control characteristics met the requirements of MIL-F-8785B(ASG) and FAR Part 23 and are satisfactory.

Ground Handling Characteristics

67. The ground handling characteristics of the C-12A aircraft were evaluated throughout the conduct of these tests. In the normal mission configuration (aft cg), two people standing inside the aircraft in the vicinity of the cabin entrance caused the nose gear strut to fully extend. The cabin door ground clearance was approximately 3 inches in this configuration. To permit safe static ground clearance, utilization of an attachable tail stand was mandatory to preclude damage to the cabin door and/or ventral fin.

68. Nose wheel steering characteristics were good. Maintaining directional control during ground operations was easily accomplished (HQRS 3). Use of the "Beta range" (propeller pitch setting) of the power control lever allowed low taxi speeds and reduced braking requirements. Braking characteristics during taxi were excellent, with no fading or overheating. No difficulty was encountered when using reverse thrust to back up for short distances. Field of view from the cockpit was good during all ground and taxi operations. Within the scope of these tests, the normal ground handling characteristics of the C-12A are acceptable.

Takeoff and Landing Characteristics

69. The takeoff characteristics of the C-12A were evaluated using normal techniques (holding brakes until takeoff power was stabilized on both engines) and short field techniques (40 percent flaps). The brakes held well during application of takeoff power and simultaneous brake release was easily

accomplished. Elevator control effectiveness was such that nose wheel lift-off was easily attainable at $0.85V_S$, without undue pilot effort or exceeding the maximum pull force of 50 pounds. Using the contractor-recommended rotation speed (110 KIAS), the maximum push force of 20 pounds was exceeded. During the takeoff performance tests (para 8) a rotation speed of 95 knots was more practical and was selected as optimum. At this rotation speed the maximum push force required was well within limits. Using the short field technique (rotating at V_{MC}), the initial climb attitude was excessive, in that the forward field of view was completely obscured. This excessive nose-high pitch attitude caused mild pilot discomfort (HQRS 5).

70. Landing characteristics were evaluated using normal landing techniques and short field techniques at the conditions shown in table 3. The short field techniques differed from the normal techniques by the use of maximum reverse thrust after touchdown. Due to the wheel lockup deficiency uncovered during landing performance tests (para 10), the short field technique was modified by delaying the application of brakes until decelerating to approximately 40 knots estimated ground speed. Preselected airspeeds ranging from 85 to 110 KIAS were used during the approaches. Maintaining a precise airspeed on the approach required moderate pilot effort (HQRS 4). Full stall landings were extremely difficult to accomplish due to the aircraft's tendency to float after roundout. Directional control was easily maintained without the use of brakes. During the landings without use of reverse thrust, the application of moderate braking immediately after flap retraction resulted in the outboard wheels locking up and causing tire blowout. This wheel lockup tendency was a deficiency and an EPR was submitted (ref 16, app A). Within the scope of these tests, the landing characteristics of the C-12A aircraft met the requirements of paragraphs 3.2.3.4 and 3.2.3.4.1 of MIL-F-8785B(ASG) but are unsatisfactory because of the brake lockup tendency. Consideration should be given to the incorporation of an antiskid device to prevent main wheel lockup.

Trim Change Characteristics

71. Trim change characteristics were evaluated at the conditions shown in table 3. The aircraft was trimmed in steady-heading balanced flight at the desired initial trim conditions and then a configuration change was made while holding one or more initial trim parameters constant. Variations in power, flap position, and gear position tested are specified in paragraphs 23.145 and 23.161 of FAR Part 23 and in paragraphs 3.6.1.2 and 3.6.3.1 of MIL-F-8785B(ASG). The quantitative test results are presented in table 13. Two peak elevator control forces resulting from configuration changes exceeded the specification limits by 4 and 5 pounds, respectively. The control forces resulting from the addition of MCP and simultaneous raising of flaps, although excessive (64 pounds), could have been easily reduced by use of the manual or electric elevator trim systems because of the length of time involved to reach those forces. During this maneuver the times required for the elevator control forces to reach 20, 40, and 64 pounds were 11, 24, and 38 seconds, respectively. All other control force variations with configuration changes were light, ranging from 10 to 25 pounds.

Table 13. Trim Change Characteristics.[1]

Specification	Pressure Altitude (ft)	Calibrated Airspeed (kt)	Landing Gear	Flaps (%)	Power Setting	Configuration Change	Parameter Held Constant	Longitudinal Control Force (lb)	
								Requirement (maximum)	Test Result
FAR Part 23: 23.145(b)(1)	10,000	130	Down	Zero	Idle	Flaps down	Airspeed	60	16
23.145(b)(2)	10,000	130	Down	Zero	Idle	Flaps up	Airspeed	60	16
23.145(b)(3)	10,000	130	Down	Zero	MCP	Flaps up	Airspeed	60	14
23.145(b)(4)	10,000	130	Down	Zero	Idle	Takeoff power	Airspeed	60	22
23.145(b)(5)	10,000	130	Down	100	Idle	Takeoff power	Airspeed	60	24
23.145(b)(6)	10,000	130	Down	100	Idle	Reduce airspeed to 110 kt and hold	Airspeed	60	20
23.145(b)(6)	10,000	130	Down	100	Idle	Increase airspeed to 144 kt and hold	Airspeed	60	12
23.145(c)	10,000	110	Up	100	PLF	Flaps up and MCP	Altitude	60	64
MIL-F-8785B(ASG) (Table XIV)	10,000	140	Up	Zero	PLF	Gear down	Airspeed and altitude	20	8
	10,000	140	Up	Zero	PLF	Gear down	Altitude	20	20
	10,000	140	Down	Zero	PLF	Flaps down	Airspeed and altitude	20	20
	10,000	14[illegible]	Down	Zero	PLF	Flaps down	Altitude	20	20
	10,000	140	Down	100	PLF	Idle power	Airspeed	20	25
	10,000	110	Down	100	PLF	Takeoff power	Airspeed	20	12
	10,000	110	Down	100	PLF	Takeoff power; gear and flaps up	Airspeed	20	12
	10,000	110	Down	Zero	Max TO	Gear up	Pitch attitude	20	18
	10,000	100	Up	100	Max TO	Flaps up	Airspeed	20	10

[1]Gross weight: 11,280 and 12,500 pounds.
Center of gravity: 181.0 inches (forward) and 196.4 inches (aft).

72. The electric trim system was objectionable due to its slow rate of travel, 49 seconds from full nose-down to full nose-up. The use of the manual trim system was preferred whenever configuration changes or large power changes were accomplished. The manual trim on all three axes became stiff when cold soaked at high altitudes (greater than 20,000 feet). The following NOTE should be added to the operator's manual:

NOTE

> The manual elevator, aileron, and rudder trim systems will become stiff during cold weather and/or high-altitude operations. Manual trim system feel will return to normal in warmer temperatures.

Night Operations

73. The night operational capability of the C-12A was evaluated during a 3.2-hour night instrument flight. The flight was conducted to evaluate all lighting systems, interior and exterior, during all phases of a typical instrument flight: taxiing and takeoff, enroute, approach, and landing.

Interior Lighting:

74. The white instrument lights were an enhancing feature, in that they significantly improved instrument readability at night with no apparent adverse effect on night vision or eye accommodation during the transition from the cockpit to outside flight environment. However, because of the presence of unlighted flight test instrumentation on the instrument panel, it was necessary to conduct most of the flight with the instrument indirect white lights ON. A similar requirement could arise operationally with the failure of one or more of the integral instrument lights. With the instrument indirect light rheostat ON, the dimming feature for the warning, caution, and advisory light panels was disabled and therefore there was no way to dim the autopilot function advisory panel lights. The advisory panel lights were too bright and throughout the flight were annoying, distracting, and severely degraded the pilot's ability to see any but the brightest objects outside the aircraft. When dimmed, the warning, caution, and advisory lights were easily seen and would be bright enough to attract the pilot's attention, regardless of the intensity of the cockpit lighting. The dimming circuit interrupt feature should be disconnected from the instrument indirect light rheostat. The inability to dim the warning, caution, and advisory panel lights with the instrument indirect light rheostat ON is a shortcoming.

75. The aircraft has eight rheostats and one switch to control all interior cockpit and instrument panel lights. The number of light switches and controls was considered excessive and consideration should be given to consolidating light controls where possible in future designs.

76. The location of the overhead floodlight is poor, limiting its usefulness for illuminating the cockpit for night ingress and egress. It could not be used as an alternate source of instrument panel illumination because it was impossible to use it to read maps and approach plates following the failure of the control yoke map light. As an emergency source of lighting, the overhead floodlight cast a glare over the entire cockpit, making it difficult to see objects outside of the airplane and difficult to read cockpit switch labels, etc. The location and design of the floodlighting system should be corrected in future designs.

77. The control yoke design was inadequate for night and instrument flight. The yokes were incompatible with any kind of knee board or device to hold the approach plates and maps. The yokes were too close to the pilot's and copilot's legs. Single maps and approach plates could be placed under the yoke but reference to them required the user to look straight down into his lap (a vertigo-inducing maneuver). Part of the map and/or approach plate was obscured by the yoke and without any way to restrain the approach plate and maps, they fell to the floor during flight in turbulence. Also, the map light provided was too bright. The light should have a rheostat instead of only an on-off switch to allow the user to adjust the light intensity to individual needs. The light could not be used at night because of the extreme glare it caused in the cockpit. Also, its intensity severely degraded the pilot's night vision adaptation. Consideration should be given to a redesign of the yoke in future designs to allow the attachment of maps or approach plates in the center of the yoke, with an indirect integral rheostat-controlled light. The clock should then be repositioned to the instrument panel where it can more easily be included in the pilot's normal instrument panel scan.

78. The cabin signs are poorly placed, difficult for the passengers to read, and do not attract the attention of the passengers in flight. The signs should be placed facing forward and aft on the cabin area fore and aft bulkheads. The poor placement of the cabin signs is a shortcoming.

79. All entrance, exit, and other cabin interior lighting is satisfactory except for the lighting for the main spar in the cabin floor. The main spar presents a dangerous obstacle in the cabin aisle because it cannot be seen when moving about the cabin in flight. The lack of illumination of the main spar in the cabin aisle is a shortcoming.

Exterior Lighting:

80. The location, position, and operation of the landing and taxi lights are excellent. The lights gave excellent runway illumination during all phases of takeoff and landing, and excellent taxiway illumination during ground operations. In addition, the location of the landing light switches next to the landing gear handle is excellent. The landing light switches are easily reached with a minimum of motion and do not require visual reference during either the takeoff or landing sequences. The location and operation of the landing light system, to include all controls, is an enhancing characteristic which should be incorporated in future designs.

RELIABILITY AND MAINTAINABILITY

81. The items listed below document problems experienced during the test program which adversely affected the reliability and maintainability of the C-12A aircraft. EPR's were submitted (refs 12 through 19, app A).

a. Poly flow tubing used in the system to sense bleed air failures vibrated loose in flight and ruptured due to excessive heat from adjacent aircraft components.

b. The propeller proximity switch in the secondary low pitch stop system failed on two occasions during the test program.

c. During maintenance daily inspection, both left-hand and right-hand inboard flap actuator attachment brackets were found to be chafing into the bottom side of the trailing edge of the wing with the flaps in the retracted position.

d. Following a No. 1 inverter failure the pilot's flight director, horizon reference, and horizontal situation indicator (HSI) became inoperative and only the HSI function could be regained.

e. During a maintenance preflight inspection the cabin door seal pressure line was found torn loose from the door seal. This is indicative of poor design and a more durable seal is required.

f. Following turns in either direction the pilot's horizon reference indicator was slow to erect to vertical. Replacement of the pilot vertical gyro produced satisfactory operation.

g. During normal flight operations the copilot altimeter indicated 100 to 200 feet higher than the pilot altimeter at all altitudes up to 30,000 feet. A subsequent failure of the altimeter set knob required instrument replacement. Changing instruments eliminated the 100 to 200-foot discrepancy between the pilot and copilot instrument readings.

82. The items listed below were shortcomings and/or system failures noted during testing for which EPR's were not submitted.

a. Autopilot: The autopilot failed to engage on the ground on numerous occasions but would engage and function properly in flight. The CMPTR flag was visible on the horizon reference indicator and the autopilot would still engage and function properly. The pitch command bar on the HSI went out of adjustment and could not be properly adjusted by BAC maintenance personnel. When the NAV mode was selected and the aircraft was already on course (CDI centered), the autopilot commanded an excessive bank angle (approximately 20 degrees) for a minute course correction. This correction resulted in a course deviation larger than the original deviation. Flight through turbulence at high airspeeds (greater than 200 KIAS) disengaged the autopilot.

b. Environmental system: On several occasions both on the ground and in flight, smoke from the environmental system filled the cockpit and cabin. In each case, the operating mode was changed and the smoke cleared in 10 minutes or less. Attempts to duplicate the same conditions which caused the smoke proved negative. On numerous occasions when the air conditioning mode was selected, the system gave unregulated heat. Similarly, when heat was selected, air conditioning often resulted. Additionally, on several flights neither heat nor air conditioning were selected, and unregulated heat resulted that could not be turned off.

c. Friction locks: After several flights the throttle friction failed to work satisfactorily, and attempts by BAC representatives to modify the system to an acceptable level were unsuccessful. If sufficient friction was applied to hold the throttles hands-off, they could not be moved to another setting without first releasing the friction.

d. Fuel system: A fuel seep was evident throughout the testing. Fuel seeped from around the integral fuel tank access plate on the bottom of the right wing. The fuel then flowed down the wing, collecting and dripping from the airspeed boom fairing. A similar access panel under the left wing also began seeping before completion of testing.

e. The transponder mode altitude transmission provided ground station readouts which varied from zero error to 500 feet higher than indicated by the pilot altimeter.

f. Landing gear system: The landing gear failed to extend after 156 flight hours and 268 landings. The landing gear motor was found to be defective and was replaced. The landing gear drive motor is normally a 5000-landing replacement item.

g. Aircraft construction: Approximately 38 rivets (27 in a row) were noted "working" in the vertical tail section, as evidenced by cracks in paint surrounding the rivet heads (including one cherry lock type rivet).

h. Annunciator system: Master warning and master caution lights illuminated frequently, with no corresponding light on the warning or caution panels. During one takeoff, the master caution light came on and was manually extinguished 14 times. The No. 2 nacelle low caution light illuminated during maneuvers where the load factor was less than +0.5. The master caution light illuminated whenever the ice vanes were extended or retracted electrically.

i. Cockpit ingress/egress: The space available to ingress/egress the cockpit was extremely limited. This resulted in an awkward entrance or exit, with clothing frequently repositioning switches. On one occasion a starter switch was inadvertently moved to the start position and the power unit was stopped to disengage the starter. Similarly, a fuel vent heater was inadvertently actuated on egress and not discovered until the fuel vent began to smoke.

j. The cabin door caution light frequently would not extinguish when the cabin door was closed and visually checked for security. Frequent adjustments were made to the microswitches but a permanent fix was not achieved.

k. The left engine fuel firewall shutoff valve occasionally failed to fully shut off fuel pressure to the No. 1 engine. Numerous actuations of the T-handle were required before proper operation was realized.

l. Toilet: During stall tests where zero g was experienced, the toilet contents were dumped into the cabin area, causing water damage to the aft cabin interior. Prior to conducting training flights executing zero g maneuvers, the toilet should be removed or drained.

CONCLUSIONS

GENERAL

83. The following conclusions were reached upon completion of the A&FC evaluation of the C-12A aircraft:

a. The C-12A aircraft met all contract guarantees, with three exceptions: (1) dual-engine cruise ceiling, (2) single-engine service ceiling, and (3) dual-engine V_H at heavy gross weight and high altitude (paras 13, 14, 15, and 17).

b. The rudder boost system greatly reduced pilot workload during asymmetric power conditions and is an enhancing feature (para 63).

c. The excellent location and operation of the landing light system, including all controls, is an enhancing feature (para 80).

d. Two deficiencies and 20 shortcomings were noted during these tests (para 6).

DEFICIENCIES AND SHORTCOMINGS

84. The following deficiencies were identified:

a. The main landing gear wheel lockup tendency during landings using brakes (para 10).

b. The lack of adequate stall warning at pressure altitudes above 20,000 feet (para 51).

85. The following shortcomings were identified:

a. Poor long-term trimmability (para 26).

b. The lightly damped and easily excited Dutch-roll oscillation with yaw damper OFF (para 41).

c. The lightening of the elevator control forces for airspeeds below trim (para 28).

d. The unsatisfactory yaw damper operation in turbulent air at airspeeds in excess of 200 KCAS (para 40).

e. The inability to dim the warning, caution, and advisory panel lights with the instrument indirect light rheostat ON (para 74).

f. The lack of constant illumination of the main spar in the cabin aisle (para 79).

g. The failure of the poly flow tubing used to sense bleed air failures (para 81).

h. The failure of the propeller proximity switch in the secondary low pitch stop system (para 81).

i. The inboard flap actuator brackets chafing into the bottom side of the trailing edge of the wing (para 81).

j. The cabin door seal pressure line tearing loose from the door seal (para 81).

k. Autopilot failure during ground check (para 82).

l. Frequent unregulated, uncommanded heat from the environmental system (para 82).

m. Improper operation of the throttle quadrant friction locks (para 82).

n. Fuel seepage around integral fuel tank access plates on both wings (para 83).

o. Rivets "working" on the vertical tail section (para 82).

p Intermittent illumination of master warning and master caution lights (para 82).

q. Frequent failure of the cabin door caution light to extinguish with door closed and locked (para 82).

r. Occasional failure of the left engine fuel firewall shutoff valve to close when the T-handle was pulled (para 82).

s. The awkward ingress and egress of the cockpit area, resulting in inadvertent actuation of overhead panel switches (para 82).

t. The dumping of the toilet contents during zero g maneuvers (para 82).

RECOMMENDATIONS

86. The deficiencies identified during this evaluation must be corrected (para 6).

87. The shortcomings should be corrected (para 6).

88. Consideration should be given to the installation of an antiskid wheel brake system (para 10).

89. If an antiskid wheel brake system is installed, additional testing should be conducted to determine the effect on landing performance (para 10).

90. Incorporate the following WARNING in the operator's manual (para 58).

WARNING

Single-engine stalls must be avoided. A high roll rate toward the dead engine will develop at stall. This roll rapidly progresses into a complete wing-over and will result in an altitude loss of 2000 feet or more prior to recovery.

91. Incorporate the following CAUTIONS in the operator's manual:

a. From paragraph 40 of this report:

CAUTION

Disengage the yaw damper during descent in turbulence at airspeeds in excess of 200 KIAS if at any time the aircraft lateral or directional oscillations begin increasing in amplitude. Disengagement of the yaw damper will allow the airplane to recover itself.

b. From paragraph 57 of this report:

CAUTION

Constant altitude turns above 20,000 feet and airspeed less than 130 KCAS may result in stall at very small bank angles.

c. From paragraph 65 of this report:

CAUTION

If a single-engine failure occurs during flight with the yaw damper engaged, high rudder pedal forces will be encountered (+200 pounds). Rapid disengagement of the yaw damper will enhance the pilot's ability to control the aircraft.

92. Incorporate the following NOTES in the operator's manual:

a. From paragraph 56 of this report:

NOTE

During practice stalls with the yaw damper engaged the pilot may experience pedal forces up to 120 pounds to maintain balanced flight. Stall recovery can be significantly improved by disengaging the yaw damper.

b. From paragraph 72 of this report:

NOTE

The manual elevator, aileron, and rudder trim systems will become stiff during cold weather and/or high-altitude operations. Manual trim system feel will return to normal in warmer temperatures.

APPENDIX A. REFERENCES

1. Regulation, Federal Aviation Administration, Federal Air Regulation. FAR Part 23, *Airworthiness Standards; Normal, Utility, and Acrobatic Category Airplanes*, 13 March 1975.

2. Letter, AVSCOM, AMSAV-EFI, 11 November 1974, subject: Test Request, U-25 (U-X) Flight Evaluation.

3. Prime Item Development Specification, Beech Aircraft Corporation, BS 22483D, "Army Model U-X and Air Force Model CX-X," 26 April 1974, revised 30 September 1974.

4. Manual, Beech Aircraft Corporation, *Operator's Manual, US Army C-12A(A)*, 30 May 1975.

5. Flight Test Manual, Naval Air Test Center, FTM No. 104, *Fixed Wing Performance*, 28 July 1972.

6. Flight Test Manual, Naval Air Test Center, FTM No. 103, *Fixed Wing Stability and Control*, 1 August 1969.

7. Handbook, Air Force Test Pilot School, FTC-TIH-70-1001, *Performance*, September 1970.

8. Handbook, Air Force Test Pilot School, FTC-TIH-68-1002, *Stability and Control*, September 1968.

9. Flight Test Manual, Advisory Group for Aeronautical Research and Development, *Volume I, Performance*, Pergamon Press, Los Angeles, California, 1959.

10. Flight Test Manual, Advisory Group for Aeronautical Research and Development, *Volume II, Stability and Control*, Pergamon Press, Los Angeles, California, 1959.

11. Military Specification, MIL-F-8785B(ASG), *Flying Qualities of Piloted Airplanes*, 7 August 1969, with Interim Amendment I, 31 March 1971.

12. Equipment Performance Report, USAAEFA, SAVTE-TA, No. 75-08-1, "C-12A Flight Evaluation," 8 September 1975.

13. EPR, USAAEFA, No. 75-08-2, 22 December 1975.

14. EPR, USAAEFA, No. 75-08-3, 22 December 1975.

15. EPR, USAAEFA, No. 75-08-4, 6 January 1976.

16. EPR, USAAEFA, No. 75-08-5, 6 January 1976.

17. EPR, USAAEFA, No. 75-08-6, 6 January 1976.

18. EPR, USAAEFA, No. 75-08-7, 8 January 1976.

19. EPR, USAAEFA, No. 75-08-8, 8 January 1976.

APPENDIX B. DESCRIPTION

GENERAL

1. The C-12A aircraft has the general structure and space arrangements of the BAC Super KingAir Model 200 aircraft. Three views of the test aircraft are shown in photos 1, 2, and 3. General specifications are listed below.

Dimensions

Wing span	54 ft, 6 in.
Horizontal stabilizer span	18 ft, 5 in.
Length	43 ft, 10 in.
Height to top of vertical stabilizer	15 ft, 0.5 in.
Propeller diameter	8 ft, 2.5 in.
Propeller/fuselage clearance	29.6 in.
Propeller/ground clearance	14.5 in.
Distance between main gear	17 ft, 2 in.
Distance between main and nose gear	14 ft, 11 in.

Cabin Dimensions

Total pressurized length	264 in.
Cabin length, partition to partition	128 in.
Cabin height	57 in.
Cabin width	54 in.
Entrance door	51.5 in. by 26.7 in.

Wing Area and Loading

Wing area	303 ft^2
Wing loading	41.3 lb/ft^2
Power loading	7.4 lb/hp

Weights

Maximum takeoff weight	12,500 lb
Maximum ramp weight	12,585 lb
Maximum landing weight	12,500 lb
Maximum zero fuel weight	10,400 lb

Photo 1. Three-Quarter Right Side View.

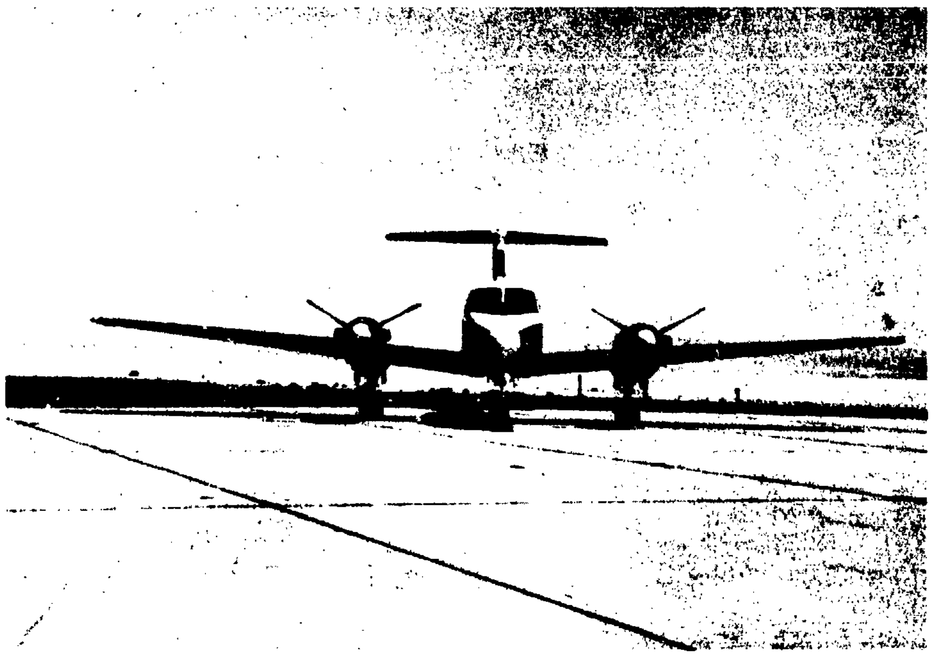


Photo 2. Front View.

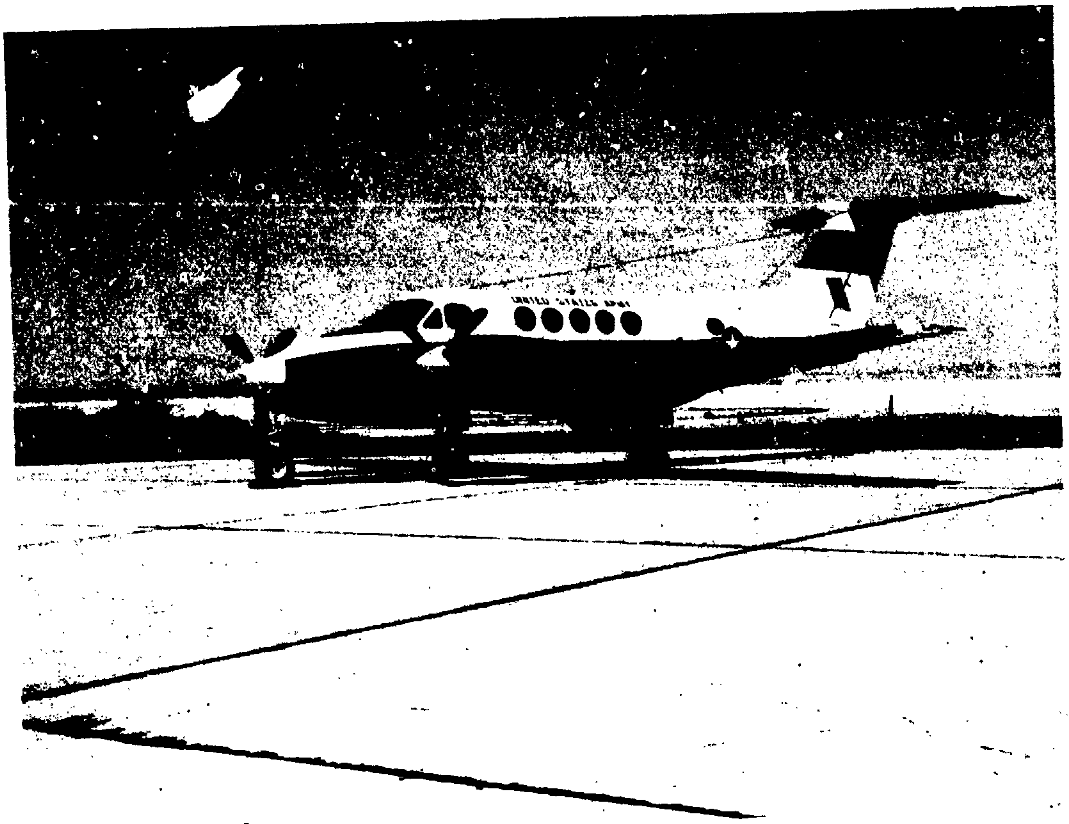


Photo 3. Left Side View.

Ground Turning Clearance

Radius for inside gear	4 ft
Radius for nose wheel	19 ft, 6 in.
Radius for outside gear	21 ft, 1 in.
Radius for wing tip	39 ft, 10 in.

FLIGHT CONTROL SYSTEM

2. The C-12A aircraft is provided with conventional dual controls for the pilot and copilot. The flight control system is reversible. The elevator and rudder control surfaces are of conventional design. The aileron control surface has a 28-inch by 1-1/2-inch metal sandwich added to the trailing edge adjacent to the trim tab to aid lateral control effectiveness. The elevators and ailerons are operated by conventional control wheels interconnected by a T-column. The rudder pedals are interconnected by a linkage below the floor. These systems are connected to the control surfaces through closed cable bell crank systems. Rudder, elevator, and aileron trim are adjustable, with controls mounted on the center pedestal. Position indicators for each of the trim tabs are integrated with their respective controls. An elevator bob weight and downspring has been incorporated to lighten longitudinal control forces in flight. A control lock is provided which permits positive locking of the control column, rudder pedals, and engine power controls.

3. A rudder boost system is provided to assist in maintaining directional control during asymmetrical thrust conditions, such as engine failure or a large variation of power between the engines. Incorporated in the rudder cable system are two pneumatic rudder boosting servos that actuate the cables to provide rudder pressure to help compensate for asymmetrical thrust. The system is operated by sensing differential pressure between each of the engine bleed air systems. The system is operated by a toggle switch located on the pedestal below the rudder trim wheel. A functional check of the system may be obtained during the conduct of normal engine run-up procedures.

4. A yaw damper system is provided to assist in maintaining directional stability. The system components include a yaw sensor, amplifier, and control valve. Regulated air pressure from the control valve is directed to the same pneumatic servos used for the rudder boost system. The system is controlled by a toggle switch adjacent to the rudder boost switch on the pedestal. The circuit of the yaw damping system is interrupted by the landing gear safety switch while the airplane is on the ground and will not operate in this condition. The system may be used at any altitude; however, it is required for flight above 17,000 feet.

ELECTRICAL SYSTEM

5. The airplane electrical system is a 28-volt direct current (VDC) (nominal) system with the negative lead of each power source grounded to the main airplane structure. DC electrical power is provided by one 34 ampere-hour, 20-cell nickel-cadmium battery and two 250-ampere starter/generators connected in parallel. The system is capable of supplying power to all subsystems that are necessary for normal operation of the airplane. A hot battery bus is provided for emergency operation of certain essential equipment and the cabin entry threshoid light circuit. Power to the main bus from the battery is through the battery relay, controlled by a security keylock switch (Army only), and a master switch, placarded BATT ON - OFF. Both are located on the overhead control panel. Power to the bus system from the generators is through generator line contactors. The voltage regulators prevent the generators from absorbing power from the bus when the generator voltage is less than the bus voltage by opening the line contactors. The generators are controlled by master switches placarded #1 GEN and #2 GEN, located on the overhead control panel.

6. Starter power to each individual starter/generator is provided from the main bus through a starter relay. The start cycle is controlled by a three-position switch for each starter, placarded IGNITION AND ENGINE START, on the overhead control panel. The starter/generator drives the compressor section of the engine through the accessory gearing. The starter/generator initially draws approximately 1100 amperes and then drops rapidly to about 300 amperes as the engine reaches 20 percent of the gas generator speed.

7. The Army aircraft has a security keylock switch, placarded OFF - ON, installed on the overhead control panel. The switch is connected into the battery relay circuit and must be ON when energizing the battery master power switch. The key cannot be removed from the lock when in the ON position.

8. For ground operation, an external power socket, located under the right wing outboard of the nacelle, is provided for the use of auxiliary power units. A relay in the external power circuit will close only if the external source polarity is correct. The security keylock switch and battery switch must be ON when applying external power. For starting, external power sources capable of up to 1000 amperes (400 amperes maximum continuous) should be used. A green advisory light on the caution/advisory annunciator panel, EXTERNAL POWER, is provided to alert the operator when the external DC power plug is connected to the airplane. Placing the avionics master power switch in the EXT PWR position will allow the use of an auxiliary power unit for avionics checkout.

ENVIRONMENTAL SYSTEM

9. The environmental system consists of the bleed air pressurization, heating and cooling system, and their associated controls. The cabin pressure vessel is designed for a normal working pressure differential of 6 psi, which will provide a cabin

pressure altitude of 3870 feet at an airplane altitude of 20,000 feet. It will provide a nominal cabin altitude of 9840 feet at an airplane altitude of 31,000 feet. A mixture of bleed air from the engines and ambient air is available for cabin pressurization at a rate of approximately 10 to 15 pounds per minute. This air mixture also passes through a heating flow control unit in each nacelle and is ducted into the cabin to provide heating. An air-to-air heat exchanger helps regulate the temperature of the bleed air. Cabin air conditioning is provided by a refrigerant gas vapor-cycle refrigeration system consisting of a belt-driven engine-mounted compressor installed in the right engine. An environmental control section on the overhead control panel provides for automatic or manual control of the environmental system.

PROPULSION SYSTEM

10. The PT6A-38 engine, manufactured by UACL, has a three-stage axial, single-stage centrifugal compressor driven by a single-stage reaction turbine. The power turbine, counterrotating with the compressor turbine, drives the output shaft. These engines are derated to produce 750 shp each under standard-day, sea-level, uninstalled conditions. Maximum continuous speed of the engine is 38,100 rpm, which equals 101.5 percent N_1. Prior to gear reduction, the turbine speed on the power side of the engine is 30,000 rpm at 2000 rpm propeller speed.

11. The Hartzell propeller is a full-feathering, constant speed, counterweighted reversing type, controlled by engine oil pressure through a single-action, engine-driven propeller governor. The propeller is three-bladed and flange-mounted to the engine shaft. Centrifugal counterweights, assisted by a feathering spring, move the blades toward the low rpm (high pitch) position and into the feathered position. Governor-boosted engine oil pressure moves the propeller to the high rpm (low pitch) hydraulic stop and reversing position.

12. The propulsion system is operated by three sets of controls: the power levers, propeller levers, and condition levers. The power levers provide control of engine power from idle through takeoff power by operation of the gas generator (N_1) governor in the fuel control unit. When the power levers are lifted over the idle detent they control engine power through the beta and reverse ranges. The propeller levers are operated conventionally and control the constant-speed propellers through the primary governor. Normal operating range is 1600 to 2000 rpm. The condition levers control the flow of fuel at the fuel control outlet and select fuel cutoff, low-idle (52 percent N_1), and high-idle (70 percent N_1) functions.

FUEL SYSTEM

13. The fuel system consists of two separate systems connected by a valve-controlled cross-feed line. Each system consists of a nacelle tank, two wing

leading edge tanks, two box section bladder tanks, and an integral (wet cell) tank, all interconnected to flow into the nacelle tank by gravity. This system of tanks is filled from the filler located near the wing tip.

14. An antisiphon valve is installed at each filler port which prevents loss of fuel or collapse of a fuel cell bladder in the event of improper securing or loss of the filler cap.

15. Each fuel system is vented through two ram vents located on the underside of the wing adjacent to the nacelle. To prevent icing of the vent system, one vent is recessed into the wing and the backup vent protrudes from the wing and contains a heating element. The vent line at the nacelle contains an in-line flame arrestor.

LANDING GEAR

16. A 28-volt split field motor, located on the forward side of the center section main spar, extends and retracts the landing gear. The landing gear motor is controlled by a switch located on the pilot subpanel which must be pulled out of detent to initiate extension or retraction. The motor incorporates a dynamic braking system, through the use of two motor windings, which prevents overtravel of the gear.

17. Torque shafts drive the main gear actuators and duplex chains drive the nose gear actuator. A spring-loaded friction-type overload clutch incorporated in the gearbox prevents damage to the structure and to the torque shafts in the event of mechanical malfunction. A 200-ampere remote circuit breaker, located on the landing gear panel forward of the main spar under the center floorboard, protects the system from electrical overload.

18. The Beech air-oil type shock struts are filled with compressed air and hydraulic fluid. Spring-loaded linkage from the rudder pedals permits nose wheel steering. When the rudder control is augmented by a main wheel brake, the nose wheel deflection can be considerably increased. As the nose wheel retracts after lift-off, it is automatically centered and the steering linkage becomes inoperative.

ANNUNCIATOR SYSTEM

19. The annunciator system consists of a warning annunciator panel (with red readout) centrally located in the glare shield and a caution/advisory annunciator panel (CAUTION yellow, ADVISORY green) located on the center subpanel. Individual function lights are of the word readout type. In the event of a fault, a signal is generated and applied to the respective channel in the appropriate annunciator panel. If the fault requires the immediate attention of the pilot, the fault warning lights on the glare shield will flash. The flashing fault warning lights may be extinguished by pressing the face of the light to reset the circuit. The

illuminated fault indication on the warning annunciator panel will remain on if the fault is not, or cannot be, corrected. If an additional fault occurs, the appropriate light on the annunciator panel will illuminate and the warning flashing light will again illuminate.

FIRE DETECTION SYSTEM

20. A fire detection system is installed to provide immediate warning in the event of fire in the engine compartments. The system consists of three photoconductive cells in each engine nacelle, control amplifiers in the center section leading edge, red warning lights in the fire control T-handles placarded #1 FIRE PULL and #2 FIRE PULL, a rotary fire protection test switch on the copilot subpanel, and a 5-ampere FIRE DETR circuit breaker panel. Flame detectors, sensitive to infrared rays, are positioned in the engine compartments to receive direct and reflected rays, thus covering the entire compartment with three cells. Heat level and rate of heat rise are not factors in the sensing method. The cell emits an electrical signal proportional to the infrared intensity and ratio of the radiation striking the cell. To prevent stray light rays from signaling a false alarm, the control amplifier activates only when the signal reaches a preset alarm level, which illuminates the appropriate warning lights in the fire control T-handles and the master fault warning light on the glare shield. When the fire has been extinguished, the cell output voltage drops below the alarm level and the control amplifier resets. No manual resetting is required to reactivate the detection system.

EMERGENCY LIGHTING SYSTEM

22. An independent battery-operated emergency lighting system is installed in the airplane. The system is actuated automatically by shock, such as a forced landing, providing adequate lighting inside and outside the fuselage to permit crew and passengers to read instruction placards and locate exits. An inertia switch, when subjected to a 2 to 3g shock, will illuminate interior lights in the cockpit, forward and aft cabin areas, and exterior lights at the overwing emergency exit and the cabin door. The battery power source is automatically recharged by the aircraft electrical system.

EMERGENCY EXIT

23. The emergency exit door, placarded EXIT-PULL, is located on the right cabin side wall just aft of the copilot seat. From the inside, the door is released with a pull-down handle and on the outside the door may be released with a flush-mounted pull-out handle. The door is of the nonhinged plug type which removes completely from the frame when the latches are released. From the inside, the door can be keylocked to prevent opening from the outside. The inside handle

will unlatch the door, whether or not it is locked, by overriding the locking mechanism. The keylock should be unlocked prior to flight to allow removal of the door from the outside in the event of an emergency. The key remains in the lock when the door is locked and can be removed only when the door is unlocked. Removal of the key from the lock before flight assures the pilot that the door can be removed from the outside if necessary.

APPENDIX C. INSTRUMENTATION

1. Instrumentation was installed in the test aircraft and maintained by USAAEFA personnel. A magnetic tape system was used as the primary means of obtaining engineering flight data. The main instrumentation package was located in the passenger cabin area at FS 198 (photo 1). An engineer flight instrument panel was also located in the passenger cabin between the crew compartment and the instrumentation package (photo 2). A pitot-static boom which incorporated angle-of-attack and angle-of-sideslip vanes was mounted on the right wing at buttline 224 (photo 3).

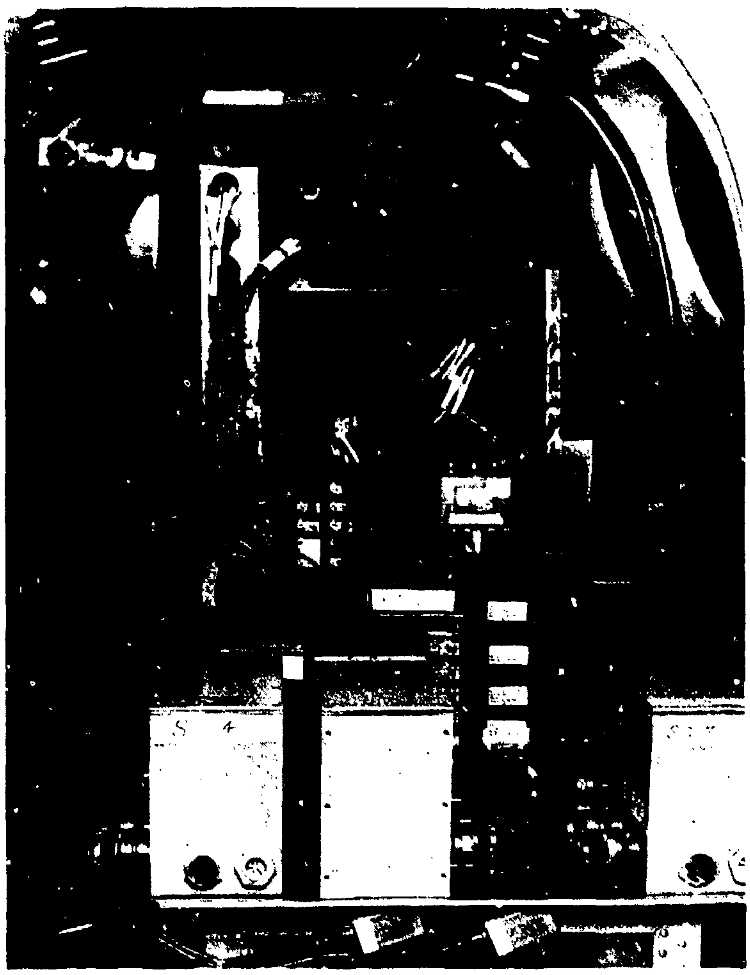


Photo 1. Instrumentation Package.

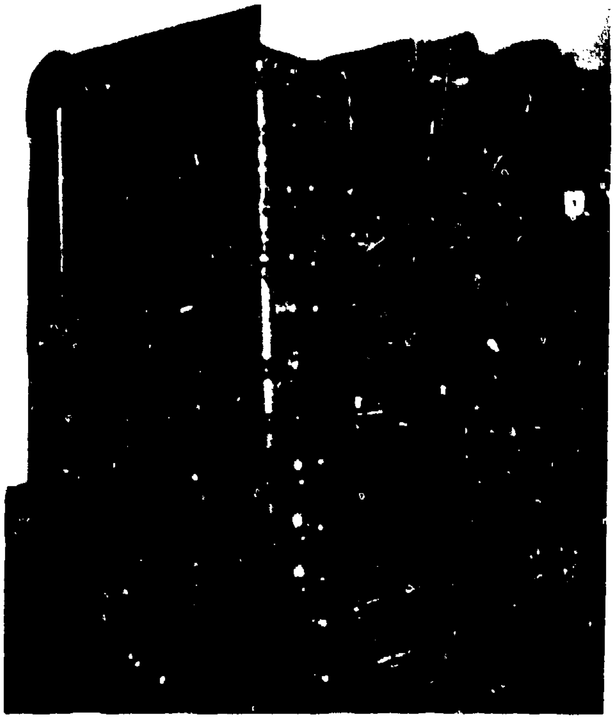


Photo 2. Engineer Panel.

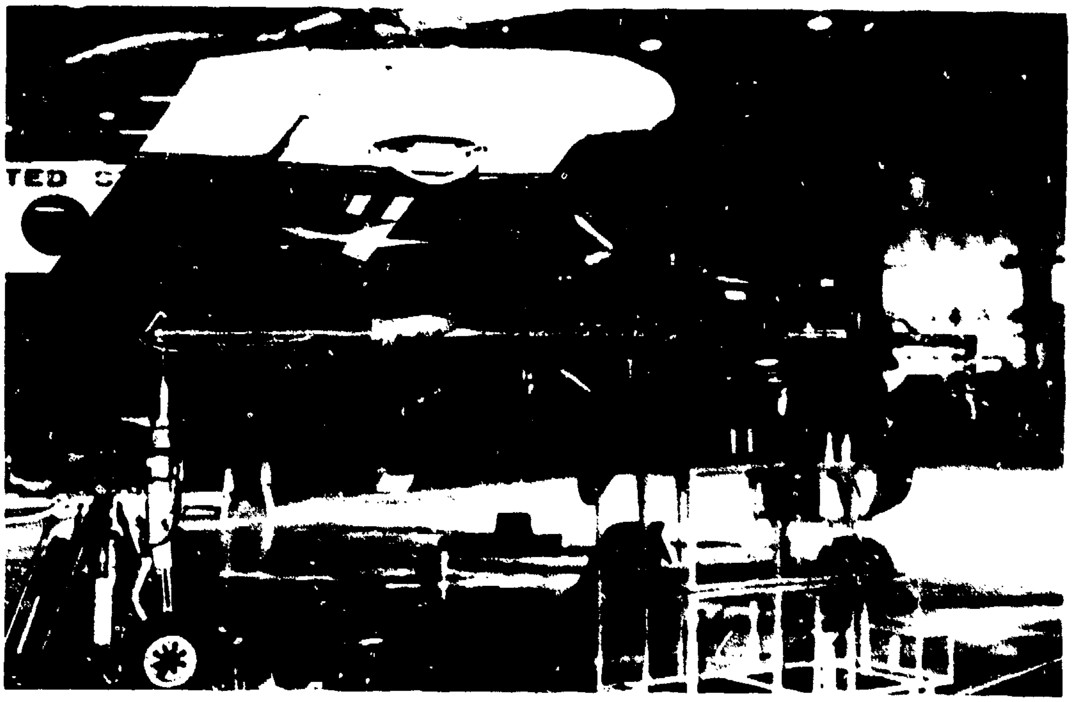


Photo 3. Pitot-Static Boom.

2. Data parameters displayed are listed below.

Pilot/Copilot Panel

Airspeed
Altitude
Vertical speed
Propeller speed, each engine
Turbine gas temperature, each engine
Center-of-gravity normal acceleration
Angle of attack

Engineer Panel

Airspeed (boom)
Altitude (boom)
Vertical speed
Outside air temperature
Propeller speed, each engine
Engine torque, each engine

3. Data parameters recorded on tape were as follows:

Airspeed (boom)
Altitude (boom)
Propeller speed, each engine
Gas producer speed, each engine
Engine torque, each engine
Turbine gas temperature, each engine
Fuel flow rate, each engine
Outside air temperature
Angle of attack
Angle of sideslip
Attitude:
- Pitch
- Roll
- Yaw

Rate:
- Pitch
- Roll
- Yaw

Acceleration:
- Center-of-gravity normal
- Center-of-gravity longitudinal
- Center-of-gravity lateral

Control position:
- Longitudinal
- Lateral
- Rudder (pedal)

Control force:
- Longitudinal
- Lateral
- Rudder (pedal)

Elevator position
Right aileron position
Rudder position
Fuel temperature,
each engine

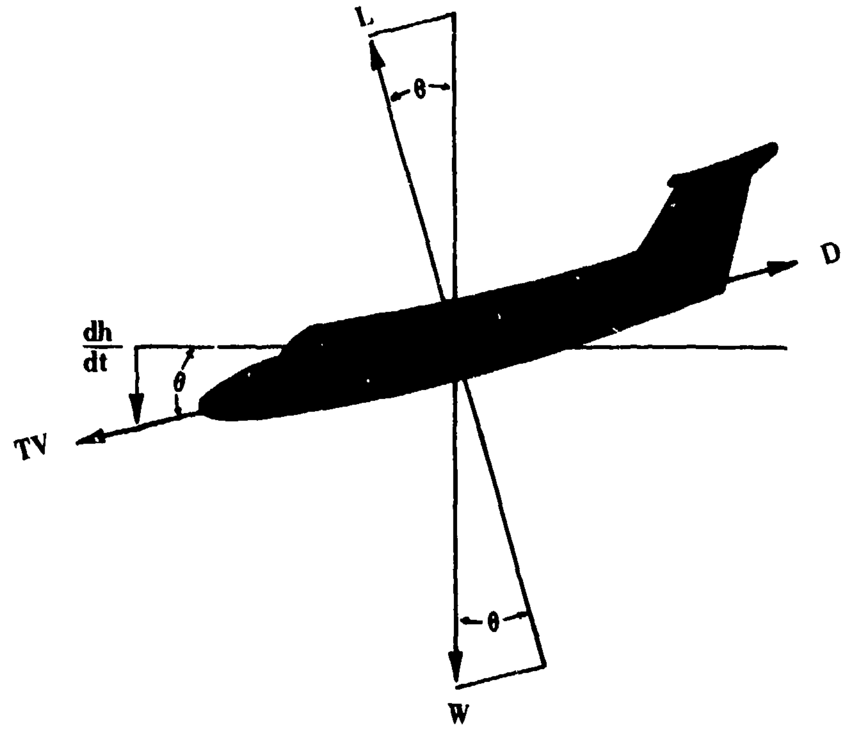
APPENDIX D. TEST TECHNIQUES AND DATA ANALYSIS METHODS

GENERAL

1. This appendix contains some of the data reduction techniques and analysis methods used to evaluate the C-12A aircraft. Topics discussed include glide, level flight and climb performance, takeoff and landing performance, airspeed calibration, and weight and balance.

GLIDE, LEVEL FLIGHT, AND CLIMB PERFORMANCE

2. The propeller-feathered glide method was used to define the base-line (minimum) drag of the C-12A aircraft in the CR and TO configurations. The method involved obtaining flight data while the aircraft was stabilized in a constant-airspeed descent with the engines shut down and propellers feathered. Parameters measured included airspeed, H_P, outside air temperature, gross weight, and elapsed time. The airspeed range from $1.1V_S$ to maximum operating airspeed was investigated for a target H_P band of 9500 to 10,500 feet. The technique used to develop the baseline-drag equation is shown below.



$$L = W \cos \theta \tag{1}$$

$$D = T + W \sin \theta \tag{2}$$

$$DV_T = TV_T + WV_T \sin \theta \tag{3}$$

$$-V_T \sin \theta = dh/dt = \frac{TV_T - DV_T}{W} \tag{4}$$

Where:

L = Lift force (lb)

W = Aircraft gross weight (lb)

θ = Descent angle (deg) = $\sin^{-1} \dfrac{dHp/dt}{Vt}$

T = Net thrust (lb) = zero with propeller feathered and engine off

D = Drag force (lb) = net thrust required for flight

V_T = Aircraft true airspeed on descent path (ft/sec)

dh/dt = Tapeline rate of descent (ft/sec) = $\dfrac{dHp}{dT} \left(\dfrac{T_T}{T_s}\right)$

$\left[\dfrac{dHp}{dt} \text{ is measured}\right]$

Considering the drag and lift force equations and applying power-off glide conditions, the following relationships can be developed:

$$C_D = \frac{D}{qS} \tag{5}$$

$$C_D = \frac{W \sin \theta}{qS} \tag{6}$$

$$C_L = \frac{L}{qS} \tag{7}$$

$$C_L = \frac{W \cos \theta}{qS} \tag{8}$$

Where:

C_D = Coefficient of drag

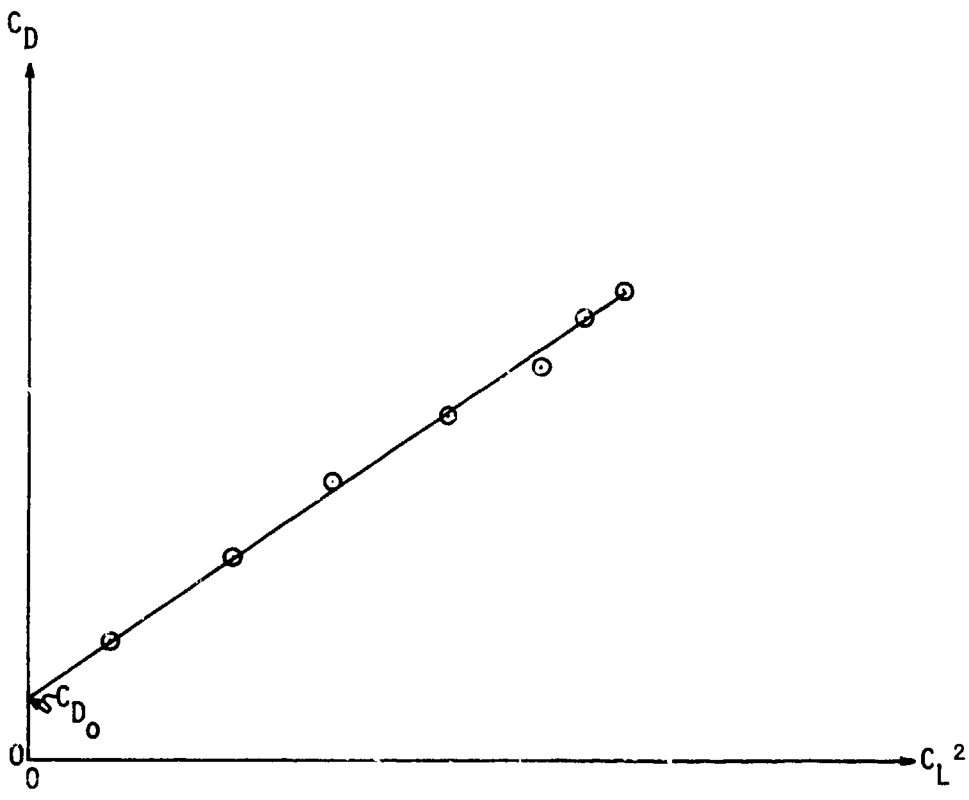
$q = 1/2\ \rho\ V^2$ (lb/ft^2) dynamic pressure

S = Wing area (ft^2)

C_L = Coefficient of lift

ρ = Air density (slug/ft^3)

The base-line drag equation ($C_{D_{BL}}$) was then developed by plotting C_D versus C_L^2 and fitting a first-order equation to the test points.



$$C_{D_{BL}} = C_{Do} + \frac{\Delta C_D}{\Delta C_L^{\,2}} C_L^{\,2} \qquad (9)$$

Figure 2.

3. During powered flight (either level flight or climbing flight), the drag of the aircraft was increased due to thrust. To reflect the change, the base-line drag equation was modified as follows:

$$\Delta C_{D_{PF-BL}} = C_{D_{PF}} - C_{D_{BL}} \tag{10}$$

Where:

$\Delta C_{D_{PF-BL}}$ = Increased drag due to thrust effect

$C_{D_{PF}}$ = Total coefficient of drag for powered flight

$C_{D_{BL}}$ = Base-line coefficient of drag

Coefficient of thrust (T_C'), thrust (T), thrust horsepower (THP), and shaft horsepower (SHP) were calculated as follows:

$$T_C' = \frac{2T}{\rho S V_T^2} \tag{11}$$

$$T = \frac{550 \times THP}{V_T} \tag{12}$$

$$THP = \eta_p \times SHP + \frac{F_n \times V_T}{550} \tag{13}$$

$$SHP = Q \times N_P \times \left(\frac{2\pi}{33{,}000}\right) \tag{14}$$

Where:

T_C' = Coefficient of thrust

T = Thrust (lb)

THP = Thrust horsepower

η_p = Propeller efficiency (obtained from propeller chart)

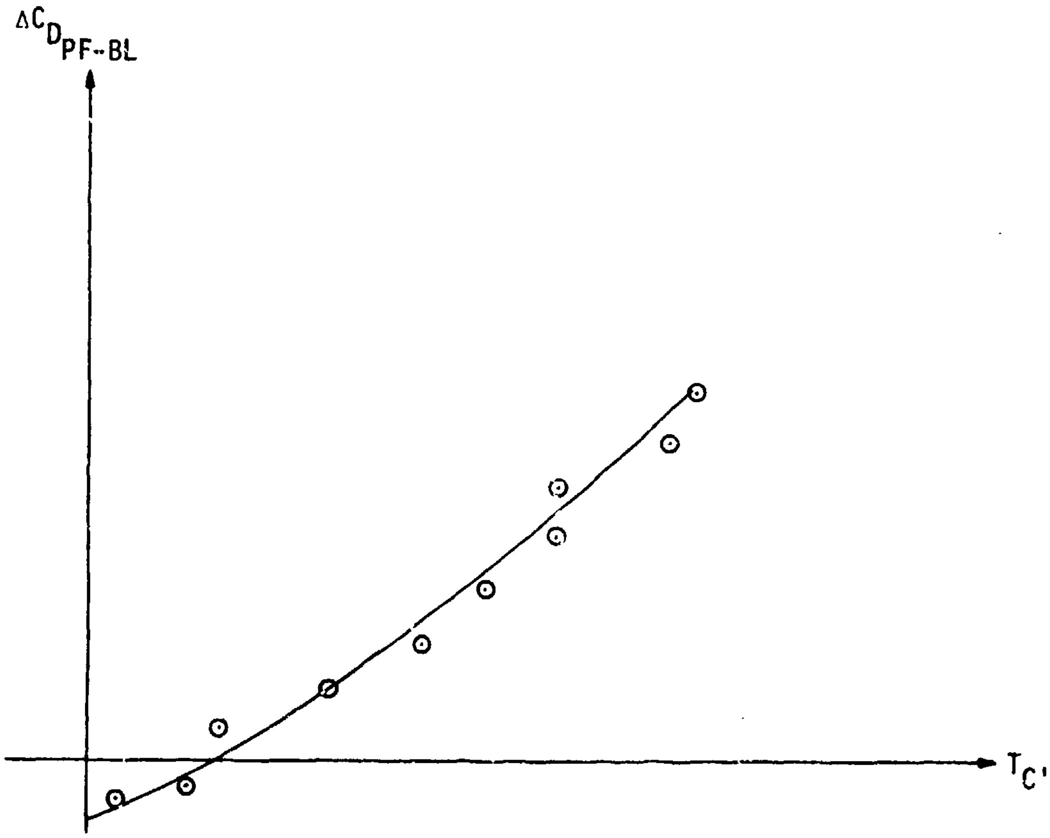
SHP = Shaft horsepower

F_n = Jet thrust (lb)

Q = Engine torque (ft/lb)

Np = Propeller speed (rpm)

The values of $\Delta C_{D_{PF\text{-}BL}}$ and T_C' were plotted to develop a generalized equation that represented the change in drag due to thrust. An equation of the second order was fitted to the data.



$$\Delta C_{D_{PF\text{-}BL}} = AT_C'^{2} + BT_C' + C \qquad (15)$$

Figure 3.

Where A, B, and C are coefficients which are constant for each flight condition.

From equation 10,

$$C_{D_{PF}} = C_{D_{BL}} + \Delta C_{D_{PF-BL}}$$

or

$$C_{D_{PF}} = C_{D_{BL}} + AT_C'^2 + BT_C' + C \qquad (16)$$

Equation 16 represents the generalized equation for all level flight and climb performance in either single- or dual-engine operation. The constant coefficients A, B, and C are tabulated in tables in the Results and Discussion section of this report. A ΔC_D of 0.001 was subtracted from the drag equation to account for the instrumentation installation.

4. Level flight performance tests (single- and dual-engine) were conducted using the constant pressure altitude method. The aircraft was stabilized and trimmed at incremental airspeeds from minimum airspeed to V_H while maintaining a constant pressure altitude. The coefficients of drag (C_D), lift (C_L), and thrust (T_C') were obtained from the recorded test data.

5. Climb performance tests (single- and dual-engine) were conducted using the sawtooth-climb method. All dual-engine climb tests were conducted with both engines operating at maximum continuous power. All single-engine climb tests were conducted with the left engine shut off and the propeller feathered while the right engine was operating at maximum continuous power. The aircraft was stabilized and trimmed at incremental airspeeds from $1.1V_S$ to $1.8V_S$ for ±1000 feet of the target altitude. The tapeline rate of climb and C_D, C_L, and T_C' were obtained from the recorded test data to determine the coefficients for the generalized equation.

6. The shp available, fuel flow rate, and net thrust of a PT6A-38 specification engine, including all installation losses, were provided by an engine computer program (standard 1518/007 dated 29 August 1972) furnished by UACL. The UACL-furnished computer deck was used to calculate the performance for an installed specification engine. The computer deck is based on the minimum performing engine that has accumulated the maximum allowable time before overhaul. For this reason, the calculated aircraft performance data, which were based on the specification engine, were always less than the observed test data. The test engines, serial numbers PC-E-79003 and PC-E-79004, used for this evaluation were production engines, each with 112 hours since overhaul as of the start of flight testing. The propeller efficiency chart was furnished by BAC and is presented in table 1. The installation losses were furnished by the BAC PIDS and are presented in table 2.

3 BLADE HARTZELL PROPELLER

ACTIVITY FACTOR = 120

C_p	J: .3	.4	5	.6	.7	8	9	1.0	1.1	1.2	1.3	1.4	1.6	1.8	2.2	2.4	2.6	2.8
.04	.574	.662	.683	.705	.709	.708	.707	706	705	.675	.689	.674	.652	.589				
05	.563	.662	.715	.758	.777	.781	784	784	.778	.756	.723	700	.681	.590				
06	590	.650	725	766	794	809	814	.816	.812	.810	.786	758	.701	627				
07	.525	.635	.710	.763	794	.821	.824	.834	.831	.831	.824	.809	.753	.698				
08	503	.615	.698	753	792	.823	.832	.844	847	848	843	.835	807	.749				
.09	481	.598	.681	740	784	820	.831	.845	.852	.853	.852	.848	826	.786				
.10	.461	.576	666	.727	777	810	.827	.842	.853	.857	859	.859	.841	.817	.719			
.12	.417	.534	.629	.698	750	787	.814	.833	.847	857	.862	.864	.858	.843	.786	.726		
14	.383	.498	588	.665	.723	.770	.797	.821	837	.850	.859	.864	.864	.855	.818	.779	.729	
16	353	.461	554	636	.699	746	.780	.806	.827	.842	.852	.858	.867	.863	.839	.811	.771	.726
18	.324	.429	.519	.602	.670	722	762	.790	.812	.830	.844	.853	.864	.865	.849	.831	.800	760
20	299	.373	487	.570	620	.668	740	774	799	.820	.835	.845	.853	.865	.855	.843	820	783
30	.204	.275	343	.411	.489	.552	.612	667	710	.743	.773	.792	.824	.857	.858	.853	.850	.810
.40											.697	.720	.777	.809	.840	.845	.848	.845

$$C_p = \frac{SHP}{2\sigma\left(\frac{N_p}{100}\right)^3\left(\frac{D}{10}\right)^5}$$

$$J = \frac{101.28\, V_T}{N_p D}$$

SHP ~ SHAFT HORSEPOWER

σ ~ ρ/ρ_0

N_p ~ PROPELLER SPEED (RPM)

D ~ PROPELLER DIAMETER (6.208 ft)

V_T ~ TRUE AIRSPEED (KNOTS)

J ~ ADVANCE RATIO

Table 1. Propeller Efficiency.

Table 2. PT6A-38 Engine Installation Losses.

Configuration	Accessory Load	Bleed Air	Inlet Pressure Recovery
Dual-engine cruise	10 hp at or below 50°F 11.5 hp above 50°F	6.75 ppm at or below 30°F 5.50 ppm above 30°F	Schedule II Figure 105
Dual-engine maximum continuous climb	10 hp at or below 50°F 12.3 hp above 50°F	6.75 ppm at or below 30°F 5.50 ppm above 30°F	Schedule I Figure 105
Single-engine maximum continuous level flight	17.20 hp	1.4 ppm	Schedule II Figure 105
Single-engine maximum continuous climb	17.20 hp	1.4 ppm	Schedule I Figure 105

AIRSPEED CALIBRATION

7. The test boom and ship's standard pitot-static system were calibrated using the pace vehicle method to determine the airspeed position error (fig. 106, app G). Calibrated airspeed (V_{cal}) was obtained by correcting indicated airspeed (V_i) for instrument error (ΔV_{ic}) and position error (ΔV_{pc}).

$$V_{cal} = V_i + \Delta V_{ic} + V_{pc} \qquad (17)$$

8. Equivalent airspeed was used to reduce the flight test data, as it is a direct measure of the free stream dynamic pressure (q).

$$V_e = V_{cal} + \Delta V_c \qquad (18)$$

Where:

ΔV_c is the compressibility correction, $q = .00339 V_e^2$

9. True airspeeds (V_T) were determined from the test altitude air density ratio (σ) and equivalent airspeed, as follows:

$$V_T = \frac{V_e}{\sqrt{\sigma}} \qquad (19)$$

TAKEOFF AND LANDING PERFORMANCE

10. Takeoff and landing performance was evaluated at three altitudes (sea level, 2000 feet H_D, and 6000 feet H_D) using an ROI (recording optical instrument) to quantify distance and ground speed.

11. Takeoff data were corrected to standard conditions. The wind correction was the first to be applied. For winds less than 5 knots, the equation is

$$Sg_w = Sg \left(1 + \frac{V_w}{V_{To}}\right)^{1.85} \qquad (20)$$

$$Sa = Sg + V_w T \qquad (21)$$

Where:

Sg = Ground distance (ft)

Sa = Air distance (ft)

V_W = Wind velocity (ft/sec)

V_{To} = Velocity at takeoff (ft/sec)

T = Time from lift-off to 50 feet (sec)

Sg_W = Ground distance corrected for wind (ft)

Sa_W = Air distance corrected for wind (ft)

Corrections for runway slope were made with the following equation:

$$Sg_{SL} = \frac{Sg_w}{1 + \dfrac{2g\ Sg_w}{V_{TO}^{\ 2}} \sin\theta} \tag{22}$$

Where:

Sg_{SL} = Ground distance corrected for slope (ft)

θ = Runway slope (positive uphill in degrees)

g = Acceleration due to gravity - 32.1741 ft/sec^2

The combined equations for thrust, weight, and density corrections are shown below. The subscripts, t and s, refer to test data (corrected for wind and runway slope) and standard data, respectively.

$$\frac{Sg_s}{Sg_t} = \frac{\dfrac{W_s}{W_t}\dfrac{\sigma_t}{\sigma_s}}{\dfrac{2g\ Sg_t}{W_t\ V_{TO_t}^{\ 2}}\left(\dfrac{W_t}{W_s} Fn_s - Fn_t\right) + 1} \tag{23}$$

using

$$h_v = \frac{V_{50}^2 - V_{TO}^2}{2g}$$

$$\frac{Sa_s}{Sa_t} = \frac{\left[\frac{W_s}{W_t}\frac{\sigma_t}{\sigma_s} h_{v_t}\right] + 50}{(h_{v_t} + 50) + \frac{Sa_t\ Fn_s}{W_s} - \frac{Sa_t\ Fn_t}{W_t}} \tag{24}$$

Where:

Subscript s refers to standard data

Subscript t refers to test data

Sg = Ground distance (ft)

Sa = Air distance (ft)

w = Gross weight (lb)

σ = Air density ratio

g = Acceleration due to gravity (ft/sec^2)

V_{TO} = Velocity at takeoff (ft/sec)

V_{50} = Velocity at 50 feet of altitude (ft/sec)

F_n = Mean net thrust

WEIGHT AND BALANCE

12. The aircraft weight and triaxial cg were determined prior to the start of flight testing. The aircraft was weighed empty in a level condition and at six pitch angles in order to obtain the vertical as well as the longitudinal and lateral cg location. The cg for the empty test aircraft with flight test instrumentation installed was determined to be FS 189.82, water line 101.97, and buttline 2.13 inches.

APPENDIX E. NOISE LEVEL MEASUREMENTS

Noise level measurements were performed by the Air Force Environmental Health Services at Edwards Air Force Base, California. The following data were extracted from a formal report, dated 4 February 1976, submitted to USAAEFA.

1. Environmental Health Services has completed in-flight noise level measurements of the C-12A aircraft at the request of the U.S. Army Aviation Engineering Flight Activity. Measurements were taken with an octave band analyzer in various cabin locations during typical flight procedures.

2. Data:

a. The flight was made on 14 January 1976 and lasted approximately 90 minutes. Attachments 1, 2, and 3 show the data recorded for the flight. Attachment 4 shows the microphone locations and the aircraft interior configuration during the flight. The aircraft normally has seats for eight passengers, but only one seat directly behind the copilot was in place during the survey. Many of the interior wall panels and partitions had been removed and some test equipment was setting on the floor. The test equipment was not operated during this flight.

b. Altitude at cruise was 6000 feet above sea level. Cabin pressure throughout the flight was maintained constant at ground level atmospheric pressure, set prior to take-off at Edwards AFB.

c. The aircraft is powered by two Canadian Pratt and Whitney PT6A-38 free shaft turbine engines, each with 750 shaft horsepower. The following power levels were used during the flight:

(1) taxi - high idle
(2) power check - 100% power
(3) take-off - 100% power
(4) normal climb - 95% power
(5) high cruise - 90% power
(6) normal cruise - 50% power
(7) descent - idle

3. Results/Conclusions:

a. Attachments 5 and 6 are from MIL-A-8806A Acoustical Noise Level in Aircraft, 12 Sep 67. The measured noise levels are within the allowable noise levels, as shown on attachments 7 and 8. These figures are plots of the allowable octave band levels specified in MIL-A-8806A and the measured levels during short duration, high cruise, and normal cruise conditions. OAL is the overall or all-pass decibel level and A is the A-weighted decibel or dBA

level. The allowable dBA levels were calculated using the respective MIL-A-8806A octave band data. For short duration conditions this is 113.7dBA, and for normal cruise conditions it is 98.7dBA.

b. In a normal passenger mode with the seats, partitions, and interior wall panels in place, the noise levels would probably be lower in the passenger area. This equipment, however, would probably not reduce the cockpit noise levels appreciably.

c. Personnel exposure criteria for Air Force activities are covered in AFR 161-35 Hazardous Noise Exposure. These criteria are based on dBA levels. For this particular set of sound level data, Air Force exposure criteria would be as follows:

(1) Short duration conditions - the peak level measured was 100dBA in the cockpit during take-off; the allowable unprotected exposure time is 30 minutes.

(2) High cruise conditions - the peak level measured was 92dBA in the cockpit; the allowable unprotected exposure time is 120 minutes.

(3) Normal cruise conditions - the peak level measured was 85dBA in the cockpit; the allowable unprotected exposure time is 420 minutes.

Combinations of these exposure levels will reduce the allowable unprotected exposure times. Also, all of the levels measured create a speech interference problem.

ENGINEERING NOISE SURVEY	DATE 14 Jan 76

PURPOSE OF EVALUATION
In-flight evaluation of C-12A (serial no. 73-22250)

OCTAVE BAND ANALYZER	MICROPHONE	CALIBRATOR
MFG B & K	MFG B & K	MFG B & K GR
MODEL 2204/1613	MODEL 4145	MODEL B&K Pistonphone 4220 GR 1562-A
SN 328886/326245	SN 334217	SN B & K 321495 GR 11734

TEMPERATURE (Degrees F)	WIND (Direction)	VELOCITY (Mph)	RH(%)
-	-	-	-

DESCRIPTION OF NOISE ENVIRONMENT (Continue on Reverse)
Various interior locations of aircraft during ground runup, taxi, take-off, cruise, and landing

SOURCE OF NOISE

PRIMARY	SECONDARY
Twin free shaft turbine engines air turbulence	-

ILLUSTRATION OF NOISE SURVEY (Continue on Reverse)

DESCRIPTIONS	DBAP	dBA	31.5	63	125	250	500	1000	2000	4000	8000	dBC
CALIBRATION (RE 0.0002 ubar dB)												
P1 taxi	102	86	99	84	86	91	88	75	67	60	54	94
P1 power check- max power	111	99	92	98	109	107	100	88	79	67	57	111
P1 take-off		100	peak									
P1 normal climb	104	92	87	99	104	102	93	79	72	61	56	104
P1 high cruise	105	92	87	92	102	101	89	84	75	63	58	105
P2 high cruise	101	88	82	95	98	89	88	82	74	64	58	100
P3 high cruise	96	87	86	84	92	89	88	82	73	66	58	95
P1 normal cruise	99	85	84	93	89	92	83	76	68	59	56	95
P2 normal cruise	92	82	78	87	90	82	82	75	68	63	59	93

REMARKS (Include AFSC/Job Codes of Those Exposed)

P1 - position #1 - microphone centered between pilot and copilot at ear level
P2 - position #2 - microphone at mid-cabin, slightly forward of trailing edge of wing, approximately 3 ft above floor
P3 - position #3 - microphone at aft cabin door approximately 3 ft above floor
In all positions microphone was held parallel to floor and facing forward

AF FORM MAR 73 1622

ENGINEERING NOISE SURVEY

DATE: 14 Jan 76

PURPOSE OF EVALUATION: In-flight evaluation of C-12A (serial no. 73-22250)

OCTAVE BAND ANALYZER	MICROPHONE	CALIBRATOR
MFG: B & K	MFG: B & K	MFG: B & K GR
MODEL: 2204/1613	MODEL: 4145	MODEL: B&K Pistonphone4220 GR 1562-A
SN: 32[illegible]/326245	SN: 334217	SN: B & K 321495 GR 11734

TEMPERATURE (Degrees F)	WIND (Direction)	VELOCITY (Mph)	RH(%)
-	-	-	-

DESCRIPTION OF NOISE ENVIRONMENT (Continue on Reverse): Various interior locations of aircraft during ground runup, taxi, take-off, cruise and landing

SOURCE OF NOISE

PRIMARY	SECONDARY
Twin free shaft turbine engines, air turbulence	-

ILLUSTRATION OF NOISE SURVEY (Continue on Reverse)

DESCRIPTIONS	UBAP	dBA	31.5	63	125	250	500	1000	2000	4000	8000	DBAR dBC
CALIBRATION (RE 0.0002 ubar dB)												
P3 normal cruise	94	81	85	86	89	82	81	74	67	62	55	92
P1 [illegible]	94	85		85	87	92	86	77	67	61	59	94
P1 taxi *	102	72	99	86	80	74	71	63	58	55	49	97

REMARKS (Include AFSC/Job Codes of Those Exposed): *Some erratic meter operation due to low-frequency aircraft vibrations

AF FORM MAR 73 1622

ENGINEERING NOISE SURVEY

DATE: 14 Jan 76

PURPOSE OF EVALUATION: In-flight evaluation of C-12A (serial no. 73-22250)

	OCTAVE BAND ANALYZER	MICROPHONE	CALIBRATOR
MFG	Bruel & kjaer (B&K)	B & K	B & K General Radio (GR)
MODEL	2204/1613	4145	B&K Pistonphone 4220 GR 1562-A
SN	328886/326245	334217	B&K 321495 GR 11734

TEMPERATURE (Degrees F)	WIND (Direction)	VELOCITY (Mph)	RH(%)
-	-	-	-

DESCRIPTION OF NOISE ENVIRONMENT (Continue on Reverse): Pre-flight and post-flight calibration

SOURCE OF NOISE

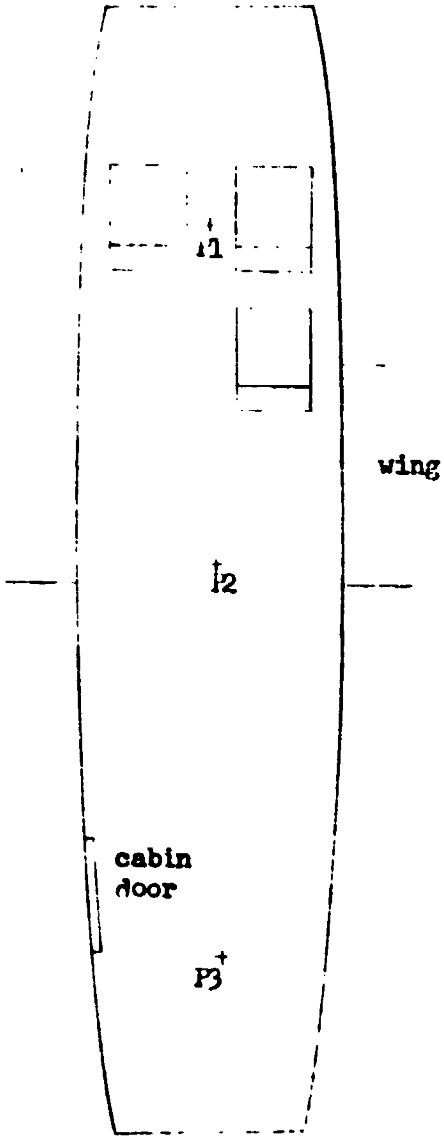
PRIMARY	SECONDARY
-	-

ILLUSTRATION OF NOISE SURVEY (Continue on Reverse)

DESCRIPTIONS	OBAP	dBA	31.5	63	125	250	500	1000	2000	4000	8000	OBAP
CALIBRATION (RE 0.0002 ubar dB)												
Pre-flight												
Pistonphone 4220						125						
GR 1562-A					114.7	114.7	114.3	114.5	113.8			
Post-flight												
Pistonphone 4220						125.2						
GR 1562-A					114.7	114.7	114.3	114.5	113.5			

REMARKS (Include AFSC/Job Codes of Those Exposed): Calibration was performed on the ground

AF FORM MAR 1622


P1
wing
P2
cabin
door
P3

3. REQUIREMENTS

3.1 Acoustical noise levels -

3.1.1 Maximum continuous power - The acoustical noise level in any part of the aircraft (see 6.2.2) intended for occupancy by the crew or other personnel shall not exceed the values specified in Table IA (preferred) or Table IB during conditions of MAXIMUM CONTINUOUS POWER.

TABLE I. - Maximum acceptable noise level at maximum continuous power

I A.				I B.		
Frequency (cps)			Max. acceptable noise level (db)	Frequency bands (cps)		Max. acceptable noise level (db)
Band		Center				
Overall			113	Overall		113
22.4 -	45	31.5	111			
45 -	90	63	111	37.5 -	75	111
90 -	180	125	111	75 -	150	111
180 -	355	250	111	150 -	300	111
355 -	710	500	105	300 -	600	105
710 -	1400	1000	99	600 -	1200	99
1400 -	2800	2000	93	1200 -	2400	93
2800 -	5600	4000	87	2400 -	4800	87
5600 -	11200	8000	87	4800 -	9600	87

3.1.2 Short duration conditions - For takeoff, afterburner operation and other conditions normally not exceeding 5 minutes continuous duration the acoustical noise level in any part of the aircraft (see 6.2.2) intended for occupancy by the crew or other personnel shall not exceed the values specified in Table II A (preferred) or Table II B.

TABLE II. - Maximum acceptable noise level under short duration conditions

II A.				II B.		
Frequency (cps)			Max. acceptable noise level (db)	Frequency bands (cps)		Max. acceptable noise level (db)
Band		Center				
Overall			120	Overall		120
22.4 -	45	31.5	118			
45 -	90	63	118	37.5 -	75	118
90 -	180	125	118	75 -	150	118
180 -	355	250	118	150 -	300	118
355 -	710	500	112	300 -	600	112
710 -	1400	1000	106	600 -	1200	106
1400 -	2800	2000	100	1200 -	2400	100
2800 -	5600	4000	94	2400 -	4800	94
5600 -	11200	8000	94	4800 -	9600	94

3.1.3. <u>Protective helmets</u> - In aircraft in which personnel must necessarily wear helmets at all times and communicate by electronic means (e.g., single place fighter aircraft), the acoustical noise level (see 6.2.2) shall not exceed the values specified in Table III A (preferred) or Table III B during conditions of MAXIMUM CONTINUOUS POWER.

TABLE III. - Maximum acceptable noise level with protective helmets or devices

III A.				III B.	
Frequency (cps)			Max. acceptable noise level (db)	Frequency bands (cps)	Max. acceptable noise level (db)
Band		Center			
Overall			113	Overall	113
22.4	- 45	31.5	111		
45	- 90	63	111	37.5 - 75	111
90	- 180	125	111	75 - 150	111
180	- 355	250	111	150 - 300	111
355	- 710	500	109	300 - 600	109
710	- 1400	1000	106	600 - 1200	106
1400	- 2800	2000	100	1200 - 2400	100
2800	- 5600	4000	94	2400 - 4800	94
5600	- 11200	8000	94	4800 - 9600	94

3.1.4 <u>Normal cruise power</u> - The acoustical noise level in any part of the aircraft (see 6.2.2) intended for occupancy by the crew or other personnel shall not exceed the values specified in Table IV A (preferred) or Table IV B, during conditions of NORMAL CRUISE POWER. Tables IV A and IV B are applicable to all Naval aircraft procurement; and to Air Force and Army aircraft procurement when so stated in the aircraft detail specification.

TABLE IV. - Maximum acceptable noise level at normal cruise power

IV A.				IV B.	
Frequency (cps)			Max. acceptable noise level (db)	Frequency bands (cps)	Max. acceptable noise level (db)
Band		Center			
Overall			106	Overall	106
22.4	- 45	31.5	104		
45	- 90	63	104	37.5 - 75	104
90	- 180	125	104	75 - 150	104
180	- 355	250	104	150 - 300	104
355	- 710	500	96	300 - 600	96
710	- 1400	1000	90	600 - 1200	90
1400	- 2800	2000	86	1200 - 2400	86
2800	- 5600	4000	75	2400 - 4800	75
5600	- 11200	8000	75	4800 - 9600	75

Figure 1. Short Duration Conditions

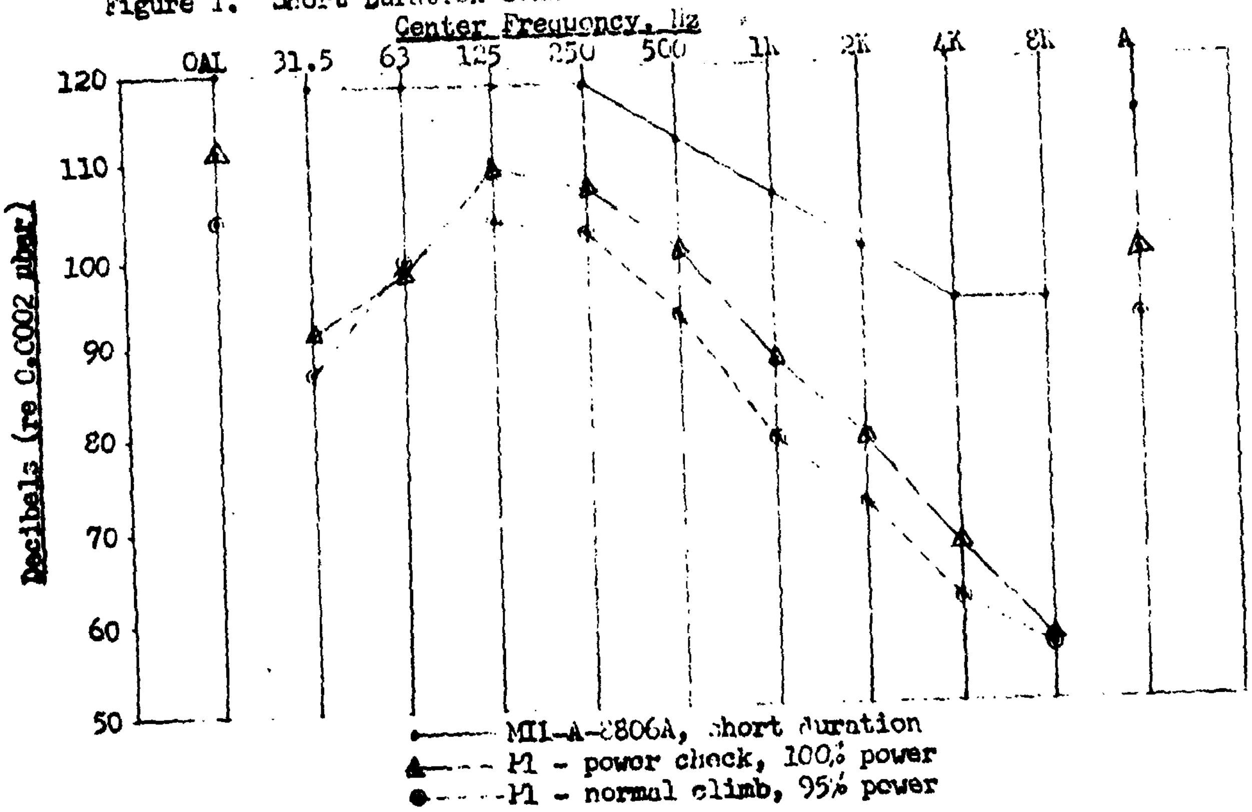


Figure 2. High Cruise Conditions

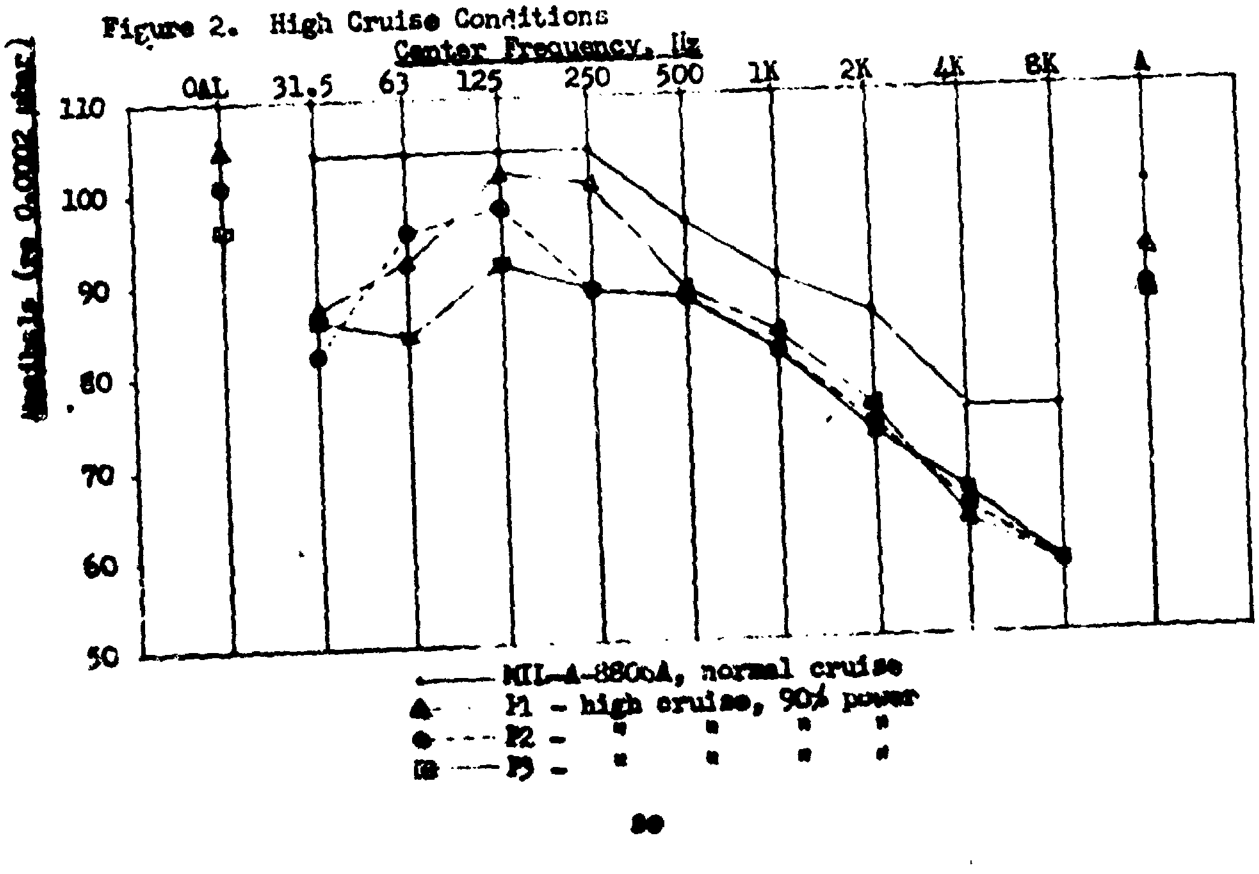
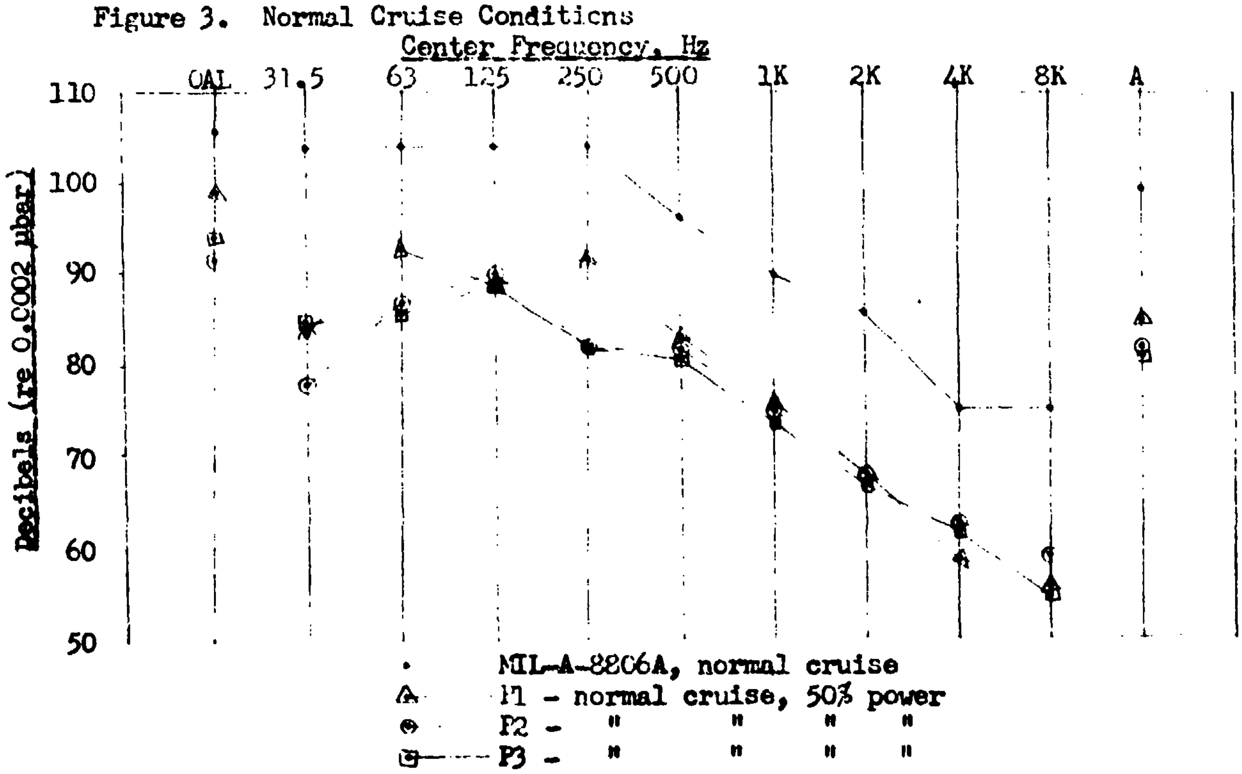
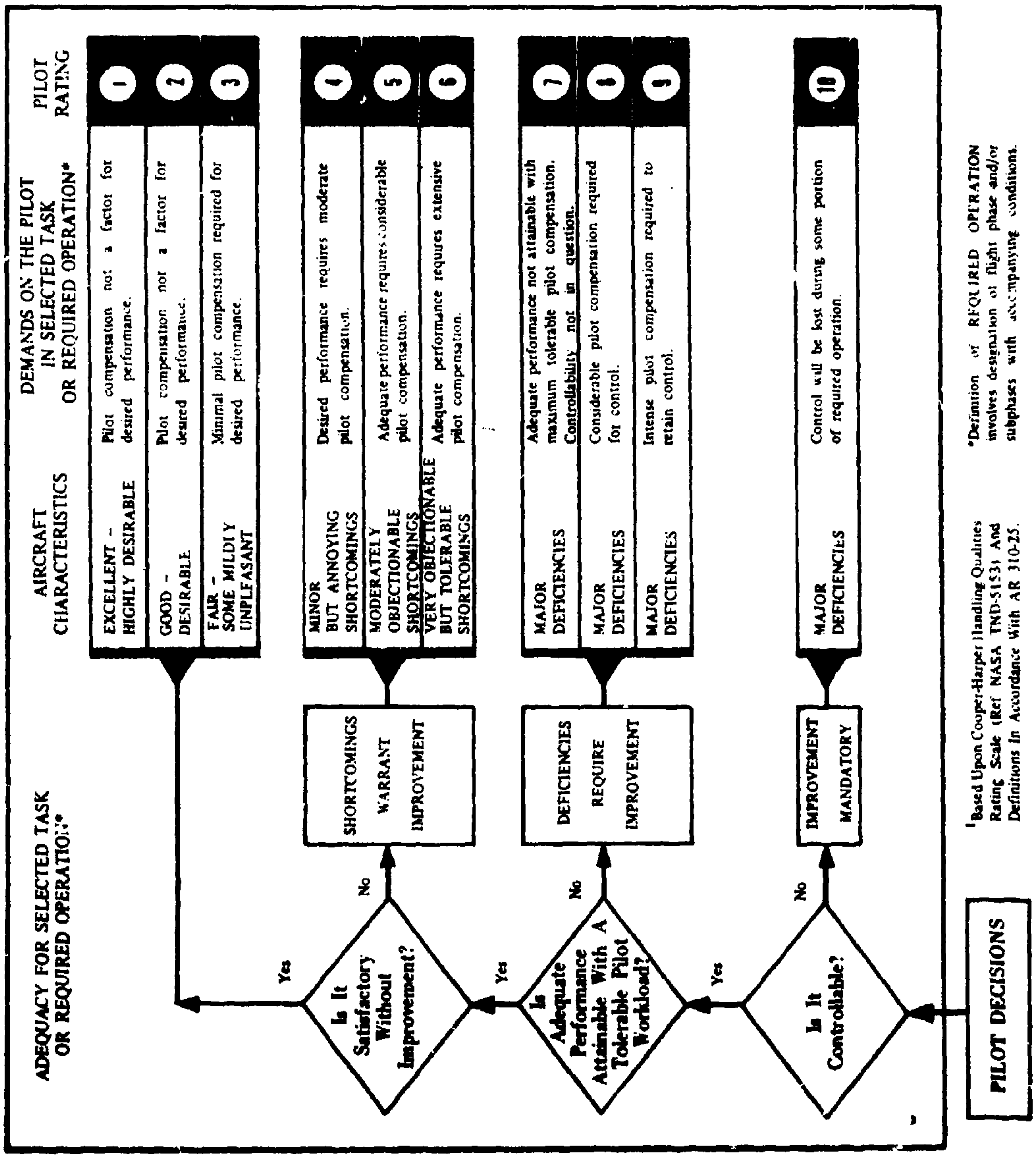


Figure 3. Normal Cruise Conditions



APPENDIX F. HANDLING QUALITIES RATING SCALE

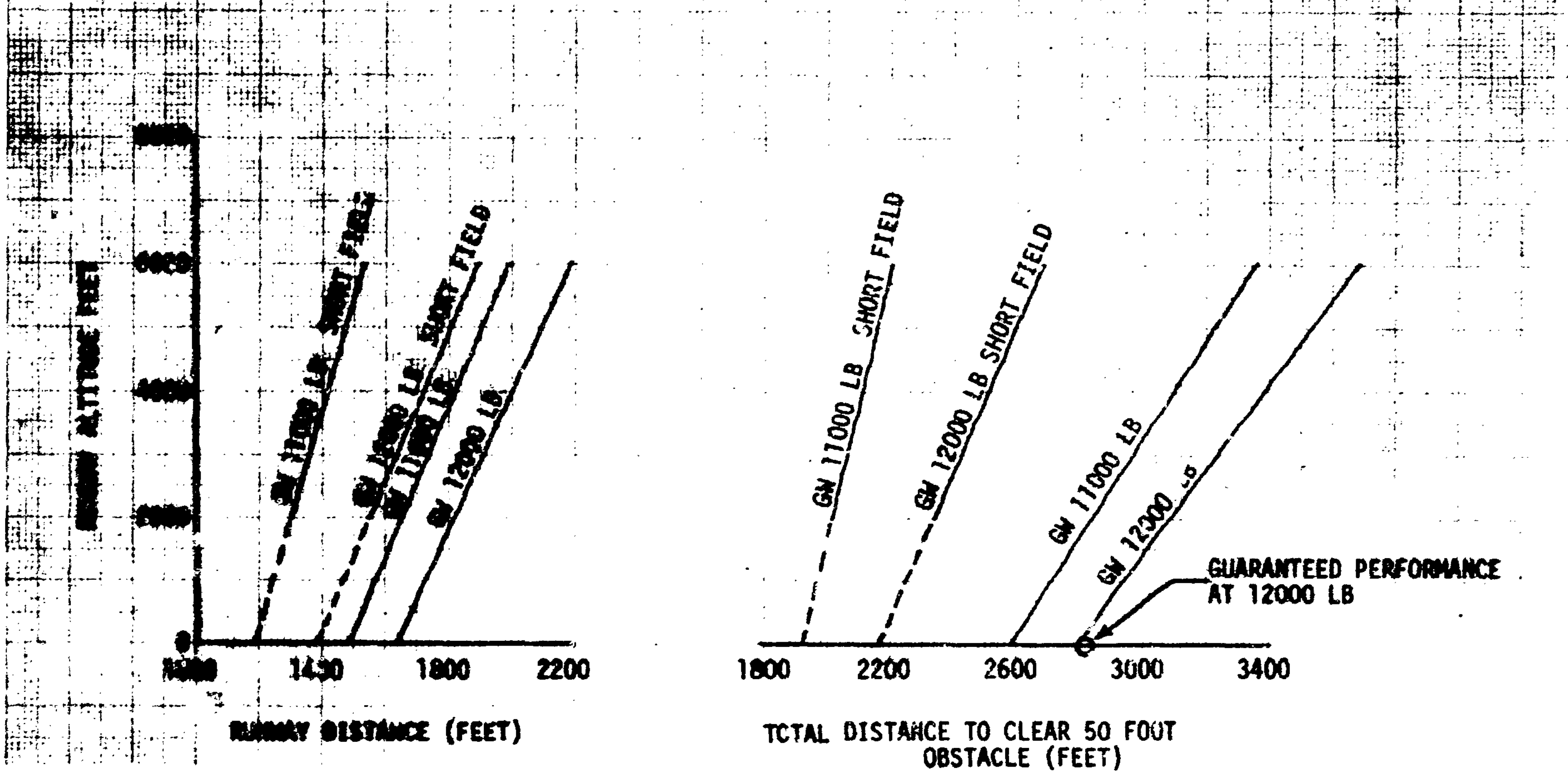


APPENDIX G. TEST DATA

INDEX

FIGURE 1
TAKE-OFF PERFORMANCE SUMMARY
C-12A USA S/N 73-22250
ENGINE MODEL PT6A-38
DRY PAVED RUNWAY
CENTER OF GRAVITY (FWD) C.G.
STANDARD DAY CONDITIONS

NOTES:
1. CURVES DERIVED FROM FIGURE 2 THRU 9.
2. SHORT FIELD TAKEOFF FLAPS = 40 PERCENT.
3. NORMAL TAKEOFF FLAPS UP.
4. PERFORMANCE BASED ON AIRSPEED PROFILE RECOMMENDED IN FIGURES 2 THROUGH 9.



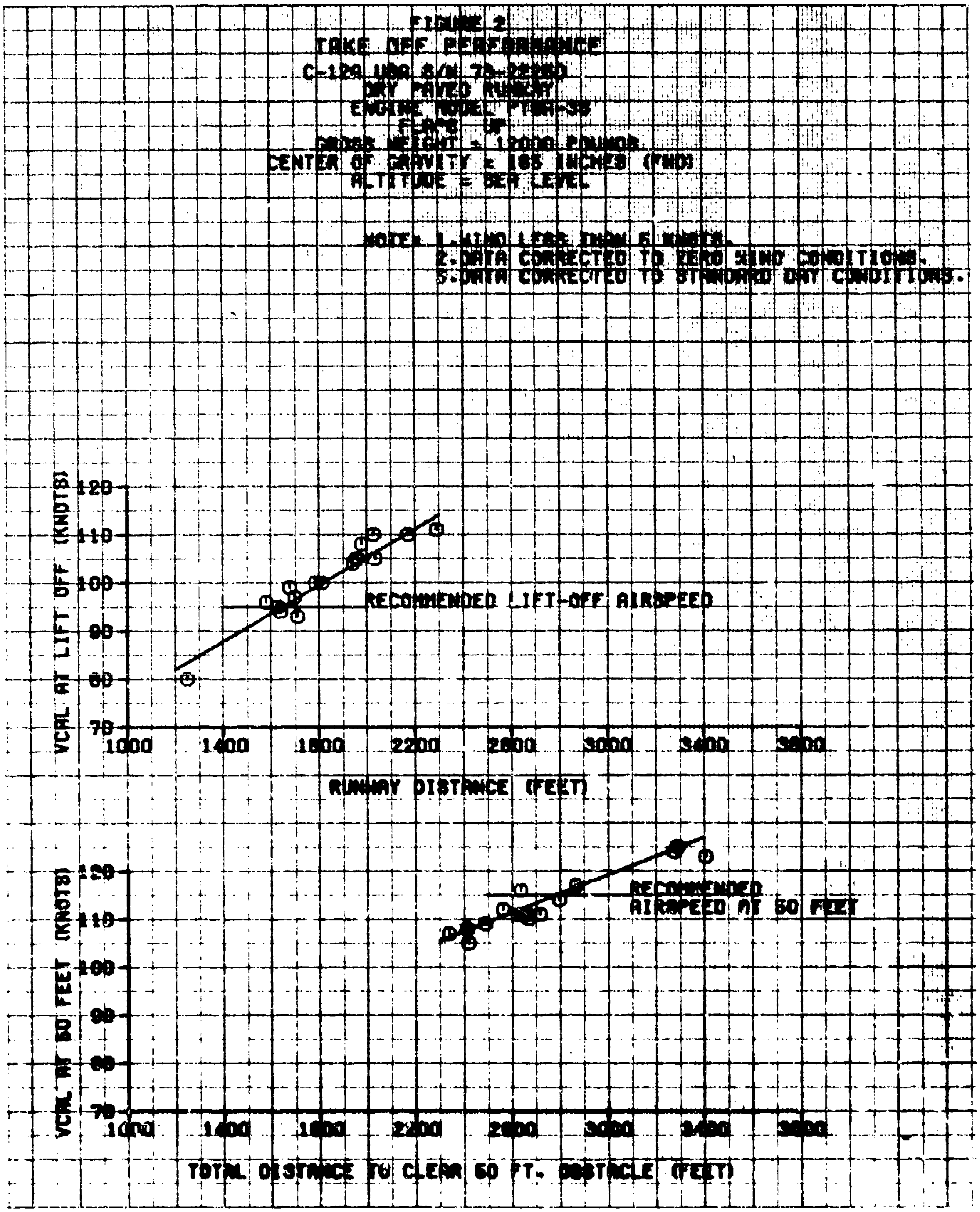


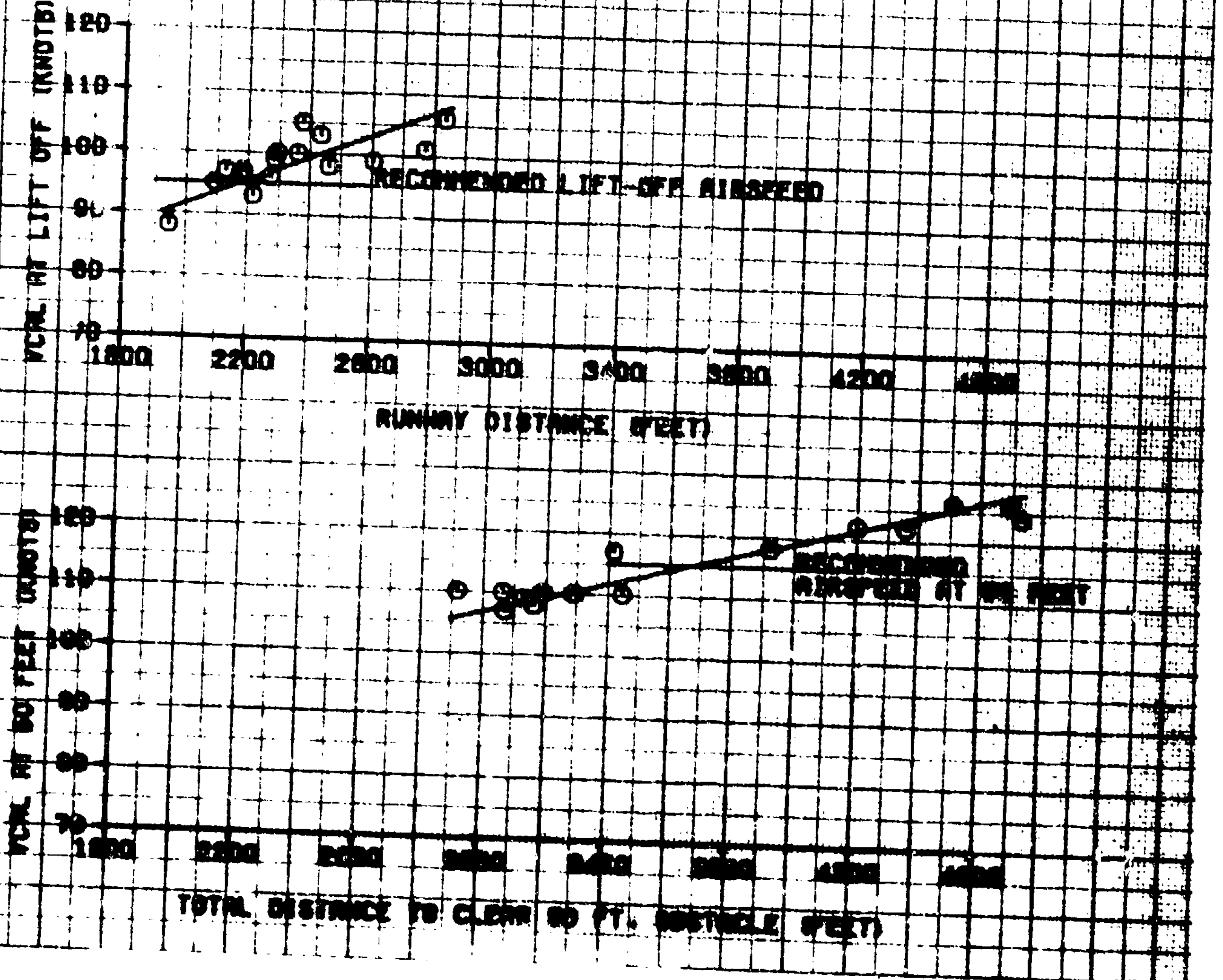
FIGURE 2
TAKE OFF PERFORMANCE
DRY PAVED RUNWAY
FLAPS UP
GROSS WEIGHT = 12000 POUNDS
ALTITUDE = SEA LEVEL
NOTE: 1. WIND LESS THAN 5 KNOTS.
2. DATA CORRECTED TO ZERO WIND CONDITIONS.
3. DATA CORRECTED TO STANDARD DAY CONDITIONS.
VCAL AT LIFT OFF (KNOTS)
120
110
100
90
80
70
1000
1400
1800
2200
2600
3000
3400
3800
RECOMMENDED LIFT-OFF AIRSPEED
RUNWAY DISTANCE (FEET)
VCAL AT 50 FEET (KNOTS)
RECOMMENDED AIRSPEED AT 50 FEET
TOTAL DISTANCE TO CLEAR 50 FT. OBSTACLE (FEET)

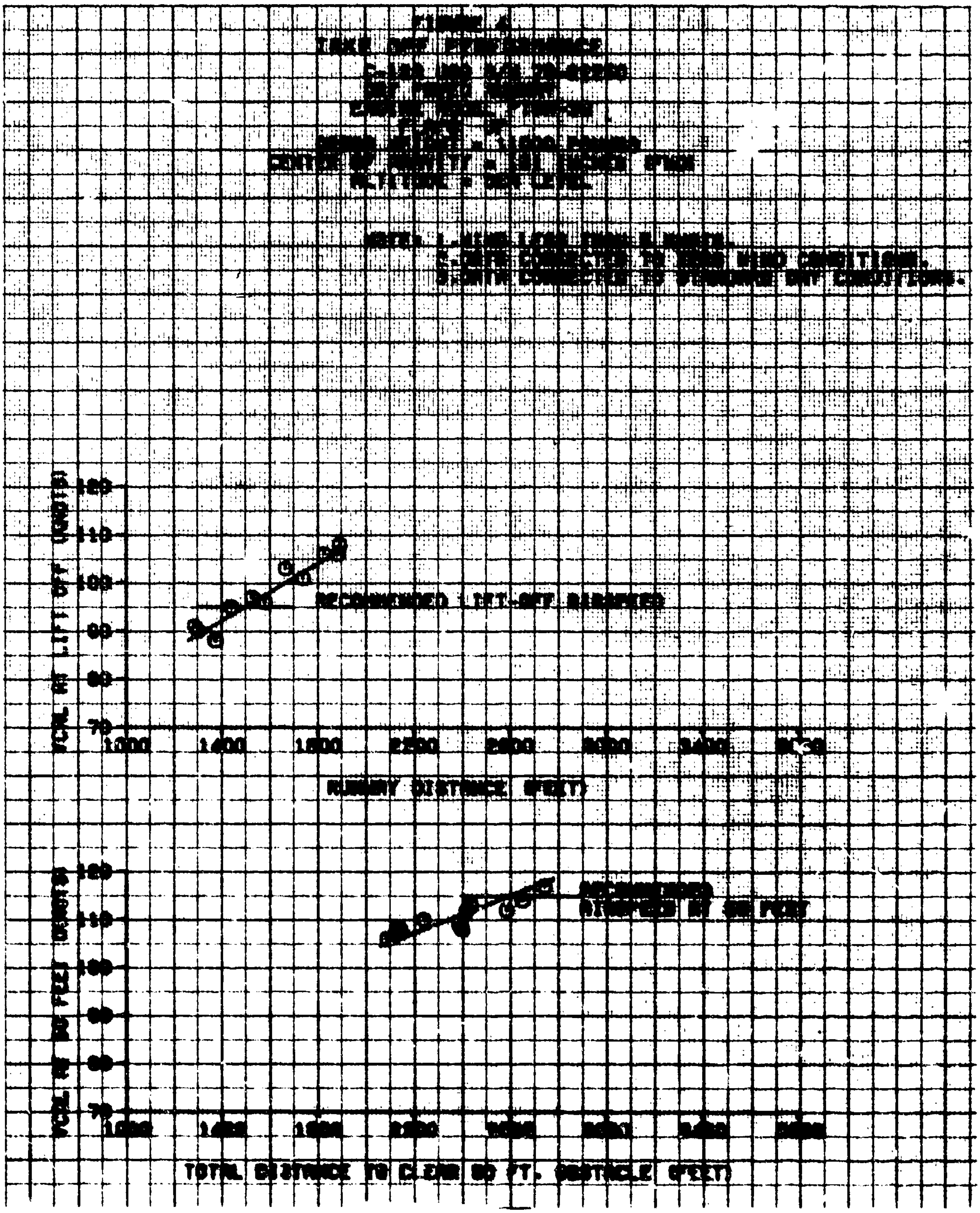
FIGURE 3

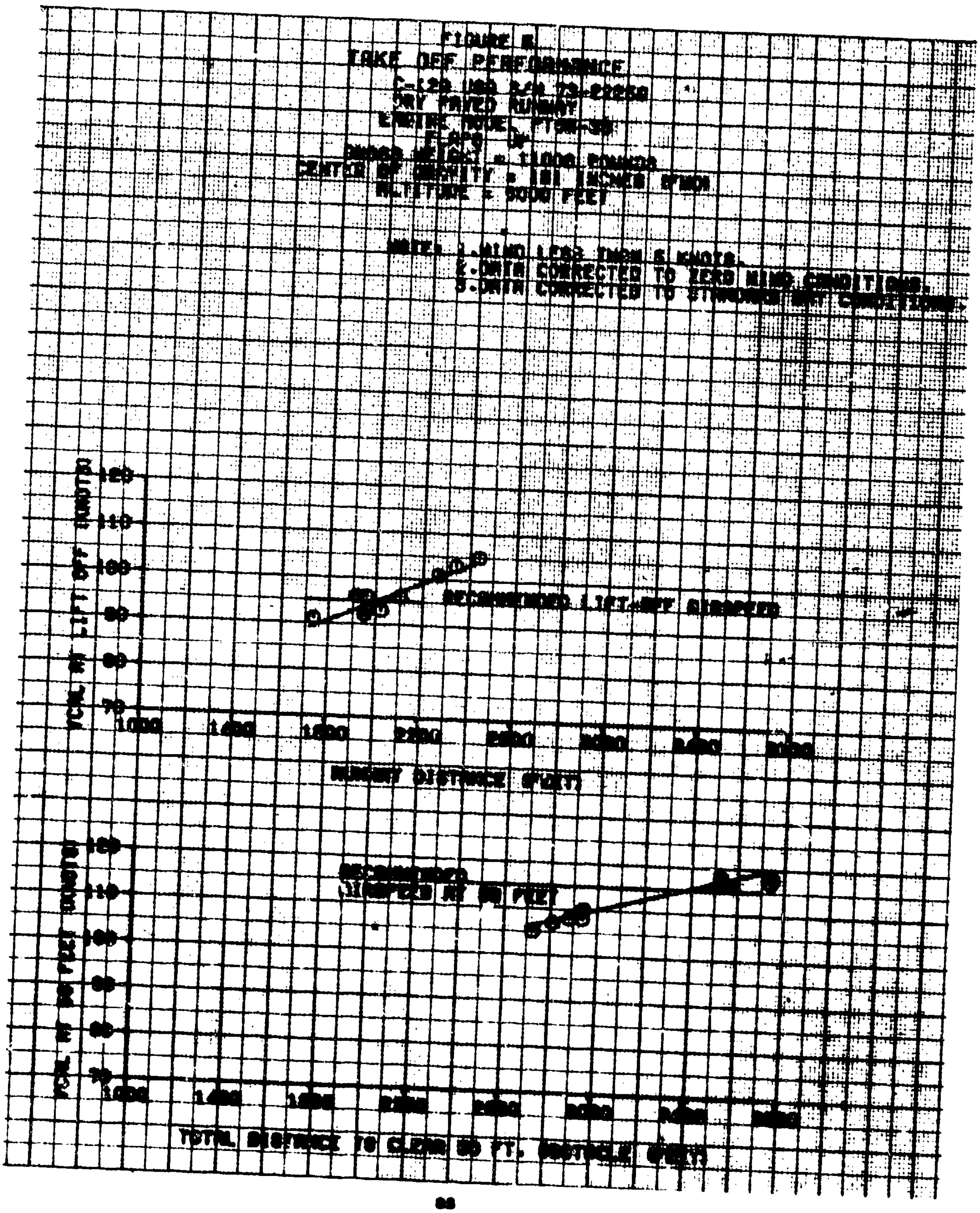
TAKE OFF PERFORMANCE

C-12A USA S/N 73-22250
DRY PAVED RUNWAY
ENGINE MODEL PT6A-38
FLAPS UP
GROSS WEIGHT = 12000 POUNDS
CENTER OF GRAVITY = 185 INCHES (FWD)
ALTITUDE = 5000 FEET

NOTE: 1. WIND LESS THAN 5 KNOTS.
2. DATA CORRECTED TO ZERO WIND CONDITIONS.
3. DATA CORRECTED TO STANDARD DAY CONDITIONS.




TAKE OFF PERFORMANCE
RECOMMENDED LIFT-OFF AIRSPEED
RUNWAY DISTANCE (FEET)
RECOMMENDED AIRSPEED AT 50 FEET
TOTAL DISTANCE TO CLEAR 50 FT. OBSTACLE (FEET)
120
110
100
90
80
70
1000
1400
1800
2200
2600
3000
3400
3800



TAKE OFF PERFORMANCE
DRY PAVED RUNWAY
2.DATA CORRECTED TO ZERO WIND CONDITIONS.
3.DATA CORRECTED TO
RECOMMENDED LIFT-OFF AIRSPEED
120
110
100
90
80
70
1000
1400
1800
2200
2600
3000
3400
3800
RUNWAY DISTANCE (FEET)
RECOMMENDED
AIRSPEED AT 50 FEET
TOTAL DISTANCE TO CLEAR 50 FT. OBSTACLE (FEET)

FIGURE [illegible]

TAKE OFF PERFORMANCE

C-12[illegible] USA S/N 7[illegible]-22[illegible]

DRY PAVED RUNWAY

ENGINE MODEL [illegible]

FLAPS [illegible] PERCENT

GROSS WEIGHT = 1[illegible] POUNDS

CENTER OF GRAVITY = [illegible] INCHES (FWD)

ALTITUDE = [illegible] FEET

NOTE: 1. WIND LESS THAN [illegible] KNOTS.
2. DATA CORRECTED TO ZERO WIND CONDITIONS
3. SHORT FIELD TECHNIQUE
4. DATA CORRECTED TO STANDARD DAY CONDITIONS

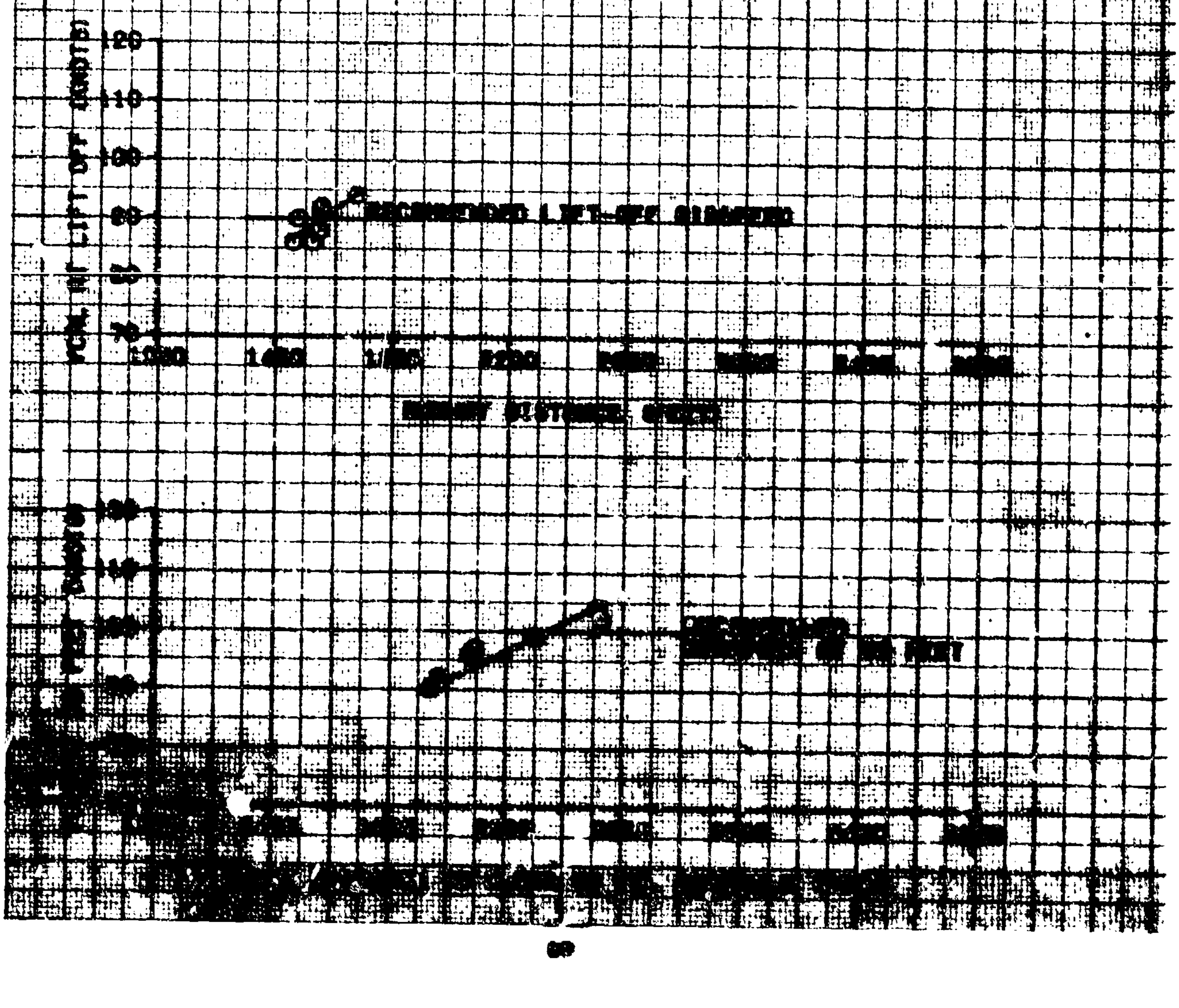
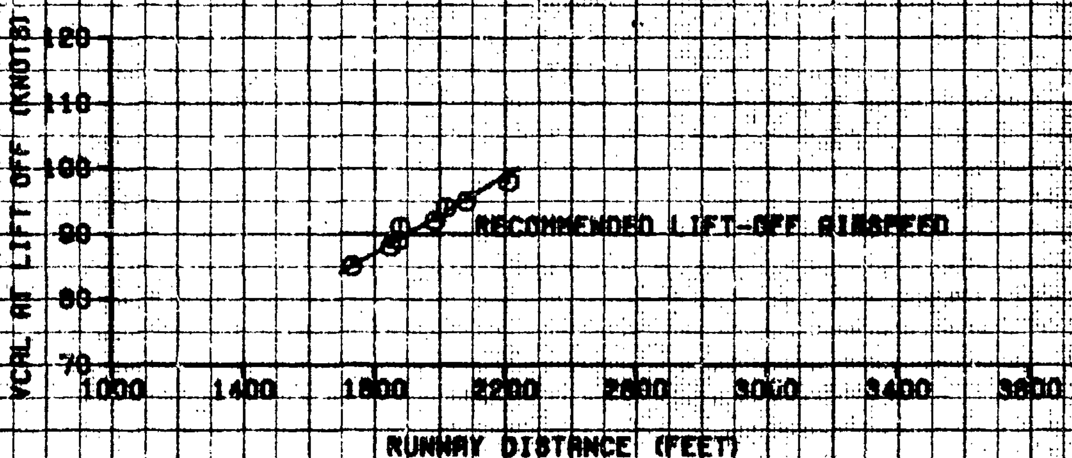
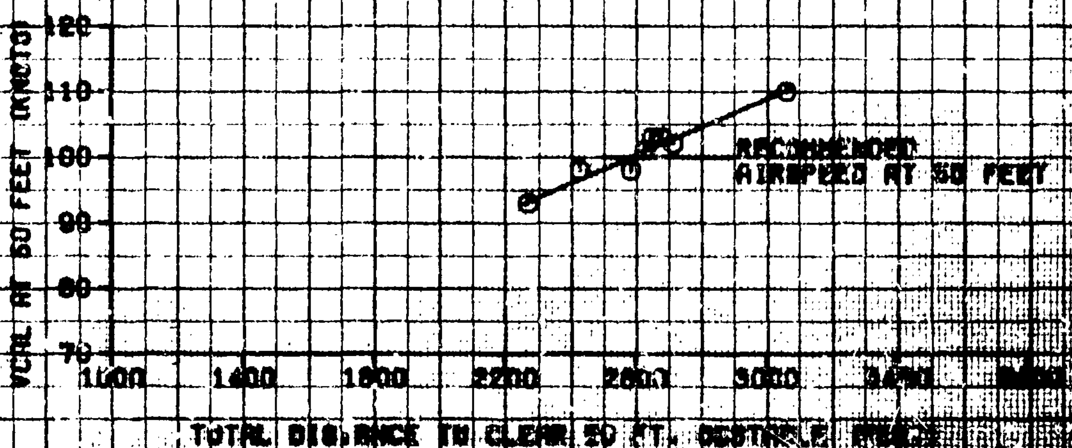


FIGURE 7
TAKE OFF PERFORMANCE
C-12A USA S/N 73-22250
DRY PAVED RUNWAY
ENGINE MODEL PT6A-38
FLAPS = 40 PERCENT
GROSS WEIGHT = 12000 POUNDS
CENTER OF GRAVITY = 185 INCHES (FWD)
ALTITUDE = 5000 FEET

NOTES: 1. WIND LESS THAN 5 KNOTS.
2. DATA CORRECTED TO ZERO WIND CONDITIONS.
3. SHORT FIELD TECHNIQUE.
4. DATA CORRECTED TO STANDARD DAY CONDITIONS.





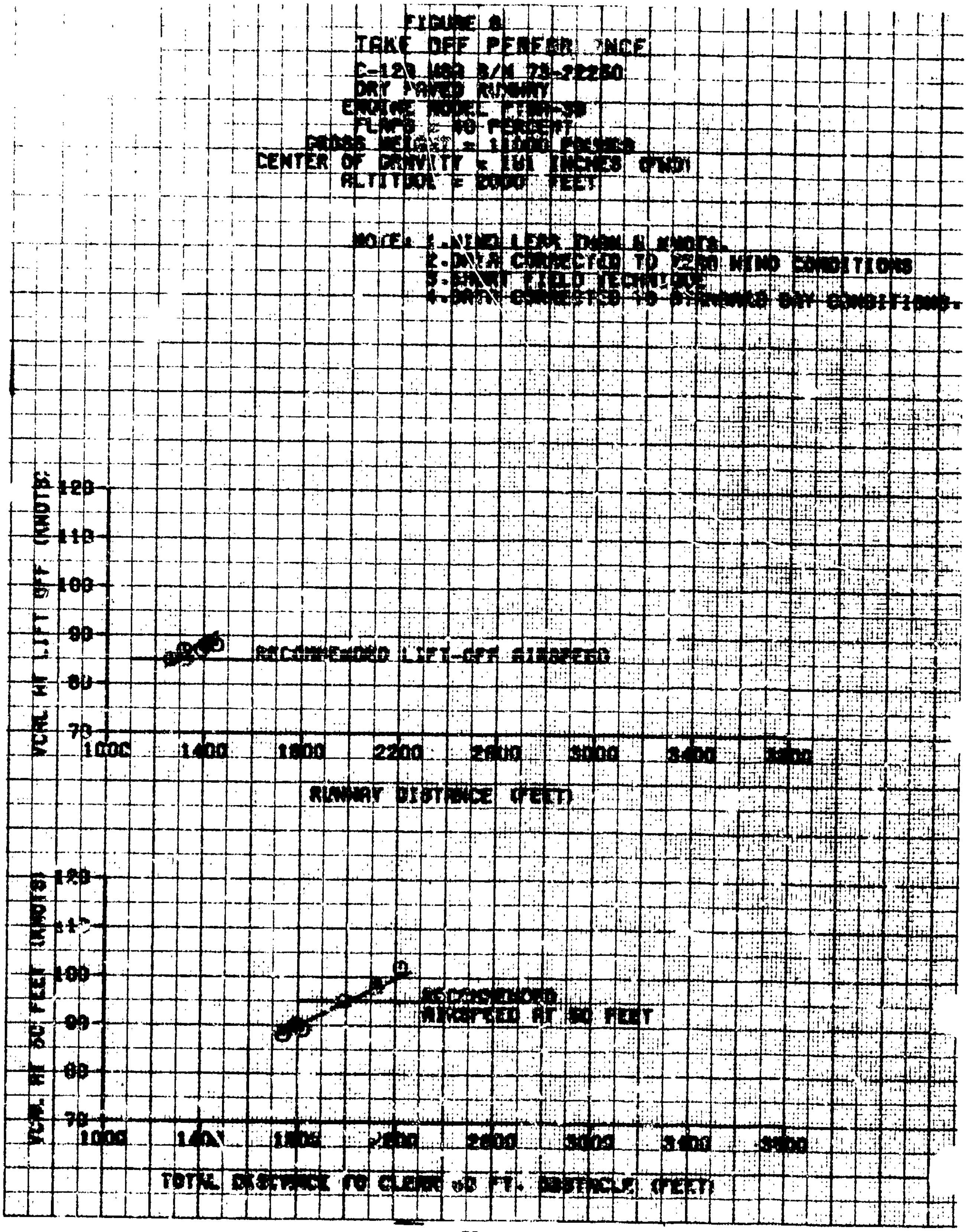


FIGURE 8
DRY PAVED RUNWAY
CENTER OF GRAVITY
120
110
100
90
80
70
1400
1800
2200
2600
3000
3400
RECOMMENDED LIFT-OFF AIRSPEED
RUNWAY DISTANCE (FEET)
120
100
90
80
70
1000
2600
3000
3400
RECOMMENDED
AIRSPEED AT 50 FEET
TOTAL DISTANCE TO CLEAR 50 FT. OBSTACLE (FEET)

FIGURE 8
TAKE OFF PERFORMANCE
C-12A USA S/N 73-22250
DRY PAVED RUNWAY
ENGINE MODEL PT6A-38
FLAPS = 40 PERCENT
GROSS WEIGHT = 11000 POUNDS
CENTER OF GRAVITY = 181 INCHES (FWD)
ALTITUDE = 5000 FEET

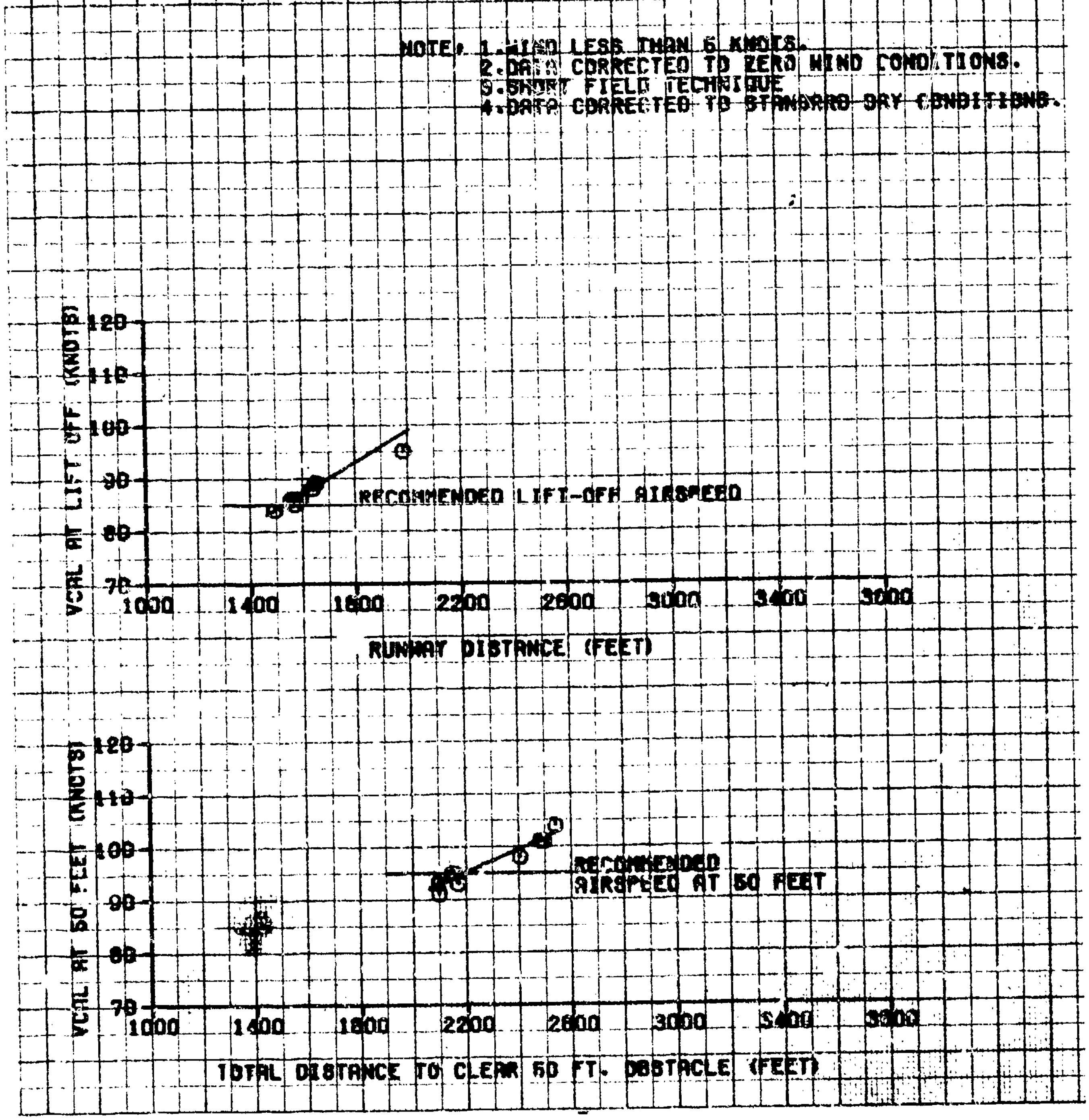


FIGURE 10
TAKE OFF PERFORMANCE
C-12A USA S/N 7?-22250
ENGINE MODEL PT6 -38

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	PROPELLER SPEED (RPM)	CONFIGURATION	FLIGHT CONDITION
12490	185.2(FWD)	500	12.1	2000	TAKE OFF	NORMAL TAKE OFF

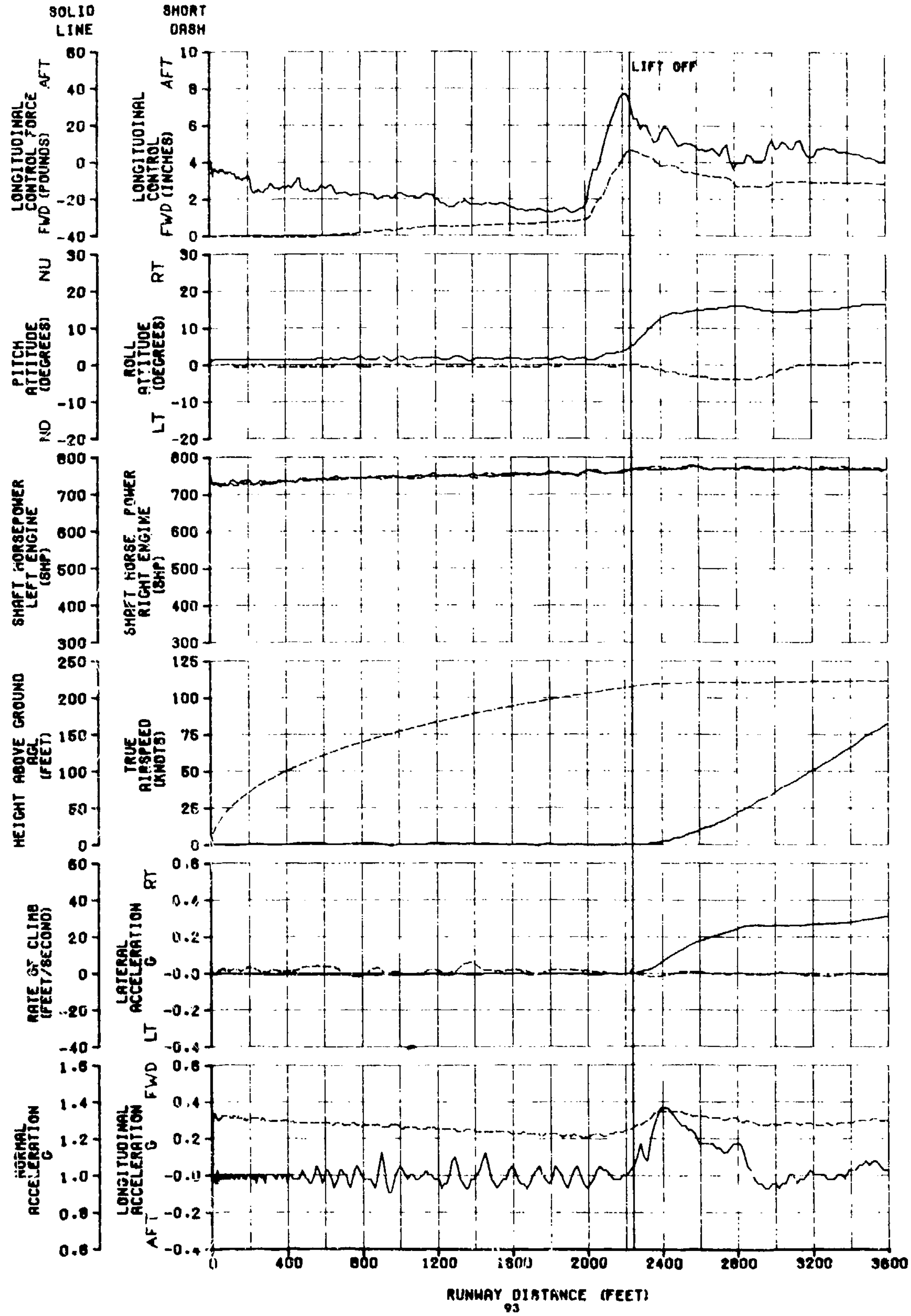


FIGURE 11

TAKE OFF PERFORMANCE

C-12A USA S/N 73-22250
ENGINE MODEL PT6A-38

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	PROPELLER SPEED (RPM)	CONFIGURATION	FLIGHT CONDITION
11380	120.7(FWD)	2510	16.4	2000	TAKE OFF	SHORT FIELD TAKE OFF

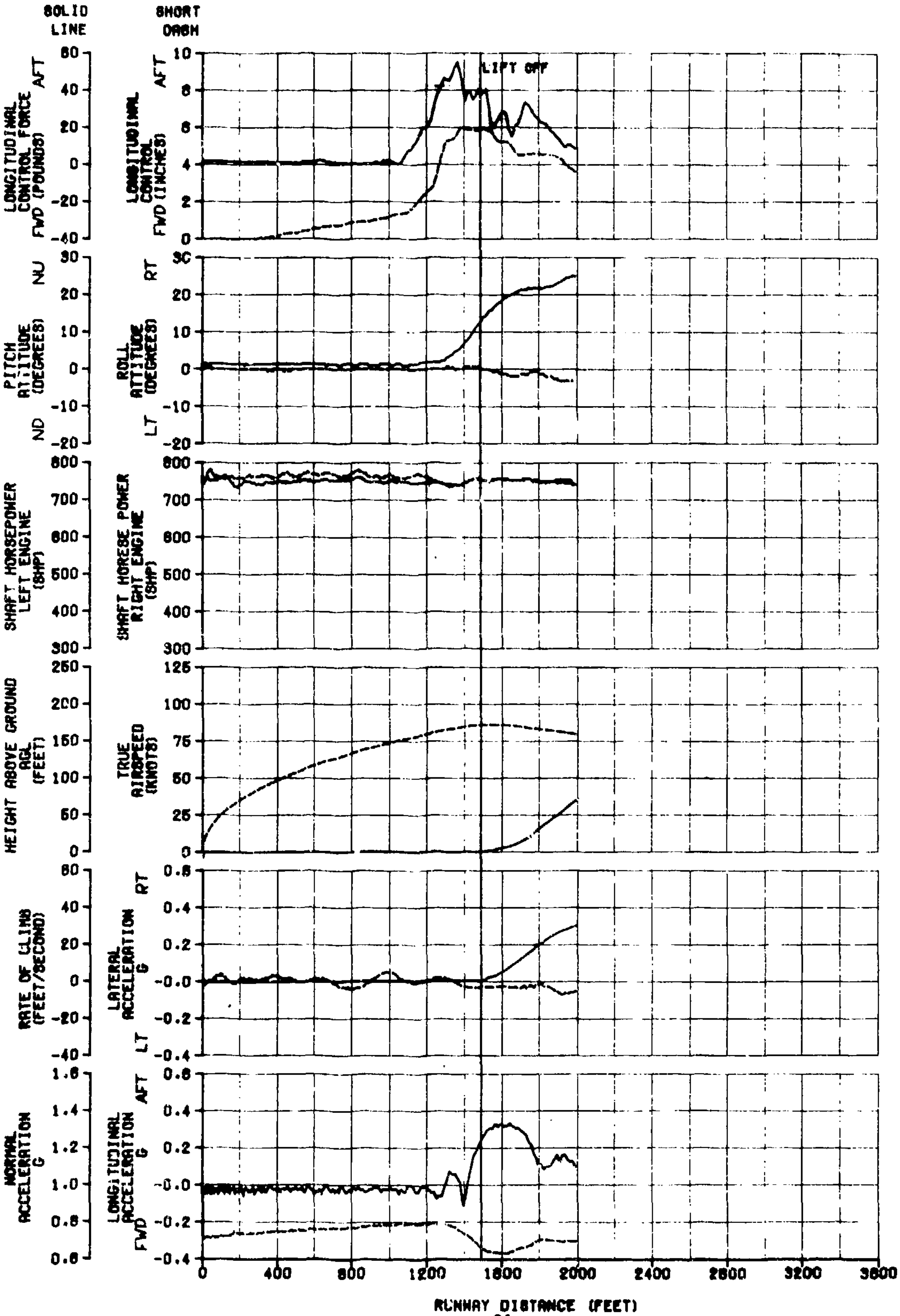


FIGURE 12
LANDING PERFORMANCE
C-12A USA S/N 73-22250
ENGINE MODEL PT6A-38
DRY PAVED RUNWAY
CENTER OF GRAVITY (FWD) C.G.
STANDARD DAY CONDITIONS

NOTES:
1. CURVES DERIVED FROM FIGURE 13 THRU 18.
2. NORMAL LANDING FLAPS = 100 PERCENT.
3. SHORT FIELD LANDING FLAPS = 100 PERCENT FULL REVERSE THRUST APPLIED AT TOUCHDOWN UNTIL 40 KIAS.
4. AIRSPEED PROFILE DURING LANDING AS SHOWN BY FIGURES 13 THROUGH 18.

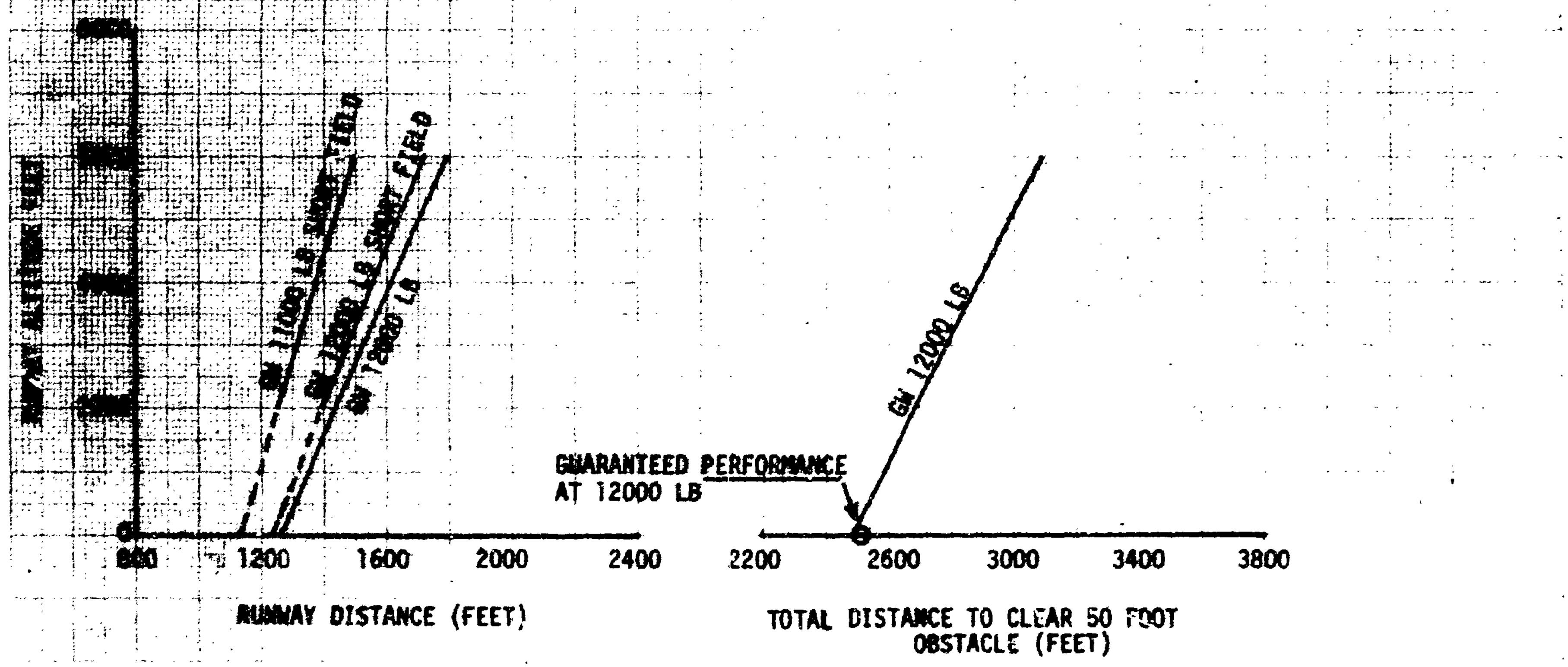
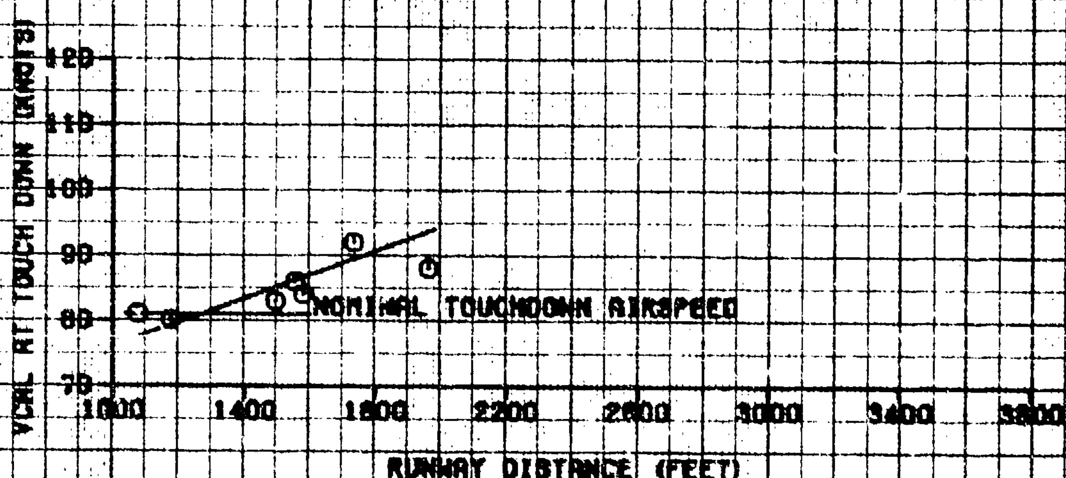


FIGURE 18
LANDING PERFORMANCE
C-12D USA S/N 73-22250
DRY PAVED RUNWAY
ENGINE MODEL PT6A-38
FLAPS = 100 PERCENT
GROSS WEIGHT = 12000 POUNDS
CENTER OF GRAVITY = 185 INCHES (FWD)
ALTITUDE = SEA LEVEL

NOTE: 1. WIND LESS THAN 5 KNOTS.
2. DATA CORRECTED TO ZERO WIND CONDITIONS.
3. DATA CORRECTED TO STANDARD DAY CONDITIONS.



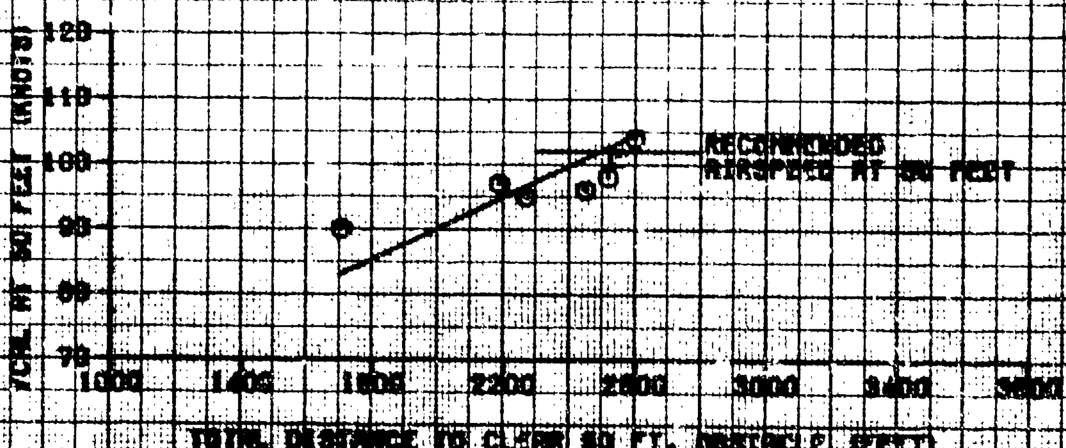


FIGURE 18

LANDING PERFORMANCE

C-12A USA S/N 73-22260
DRY PAVED RUNWAY
ENGINE MODEL PT6A-38
FLAPS = 100 PERCENT
GROSS WEIGHT = 12000 POUNDS
CENTER OF GRAVITY = 185 INCHES (FWD)
ALTITUDE = 5000 FEET

NOTE: 1. WIND LESS THAN 5 KNOTS.
2. DATA CORRECTED TO ZERO WIND CONDITIONS.
3. DATA CORRECTED TO STANDARD DAY CONDITIONS.

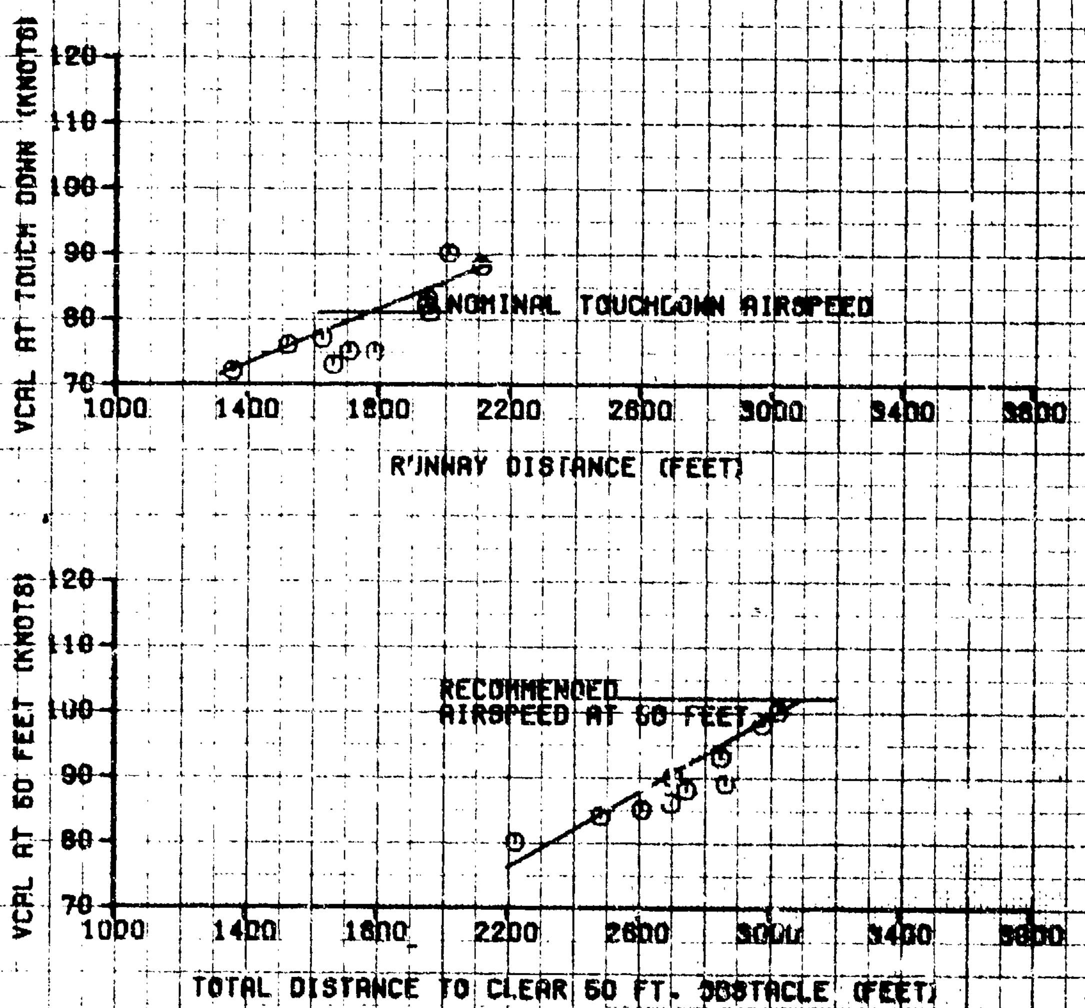
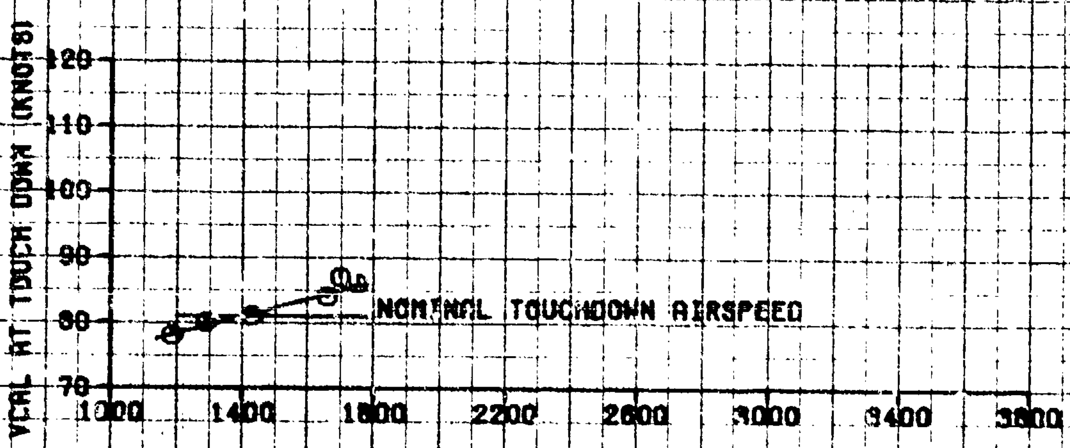


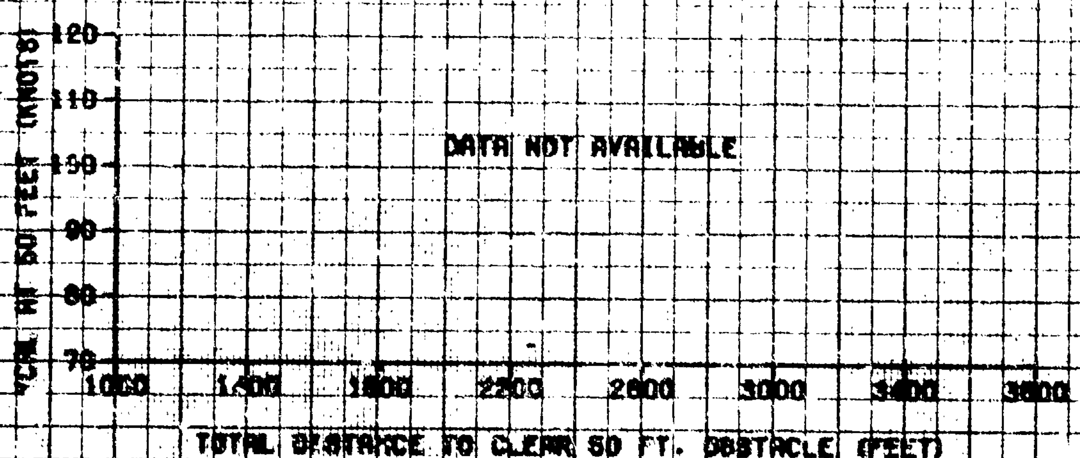
FIGURE 15

LANDING PERFORMANCE

C-12A USA S/N 73-22250
DRY PAVED RUNWAY
ENGINE MODEL PT6A-38
FLAPS = 100 PERCENT
GROSS WEIGHT = 12000 POUNDS
CENTER OF GRAVITY = 185 INCHES (FWD)
ALTITUDE = 2000 FEET

NOTE: 1. WIND LESS THAN 5 KNOTS.
2. DATA CORRECTED TO ZERO WIND CONDITIONS.
3. SHORT FIELD TECHNIQUE
4. DATA CORRECTED TO STANDARD DAY CONDITIONS.





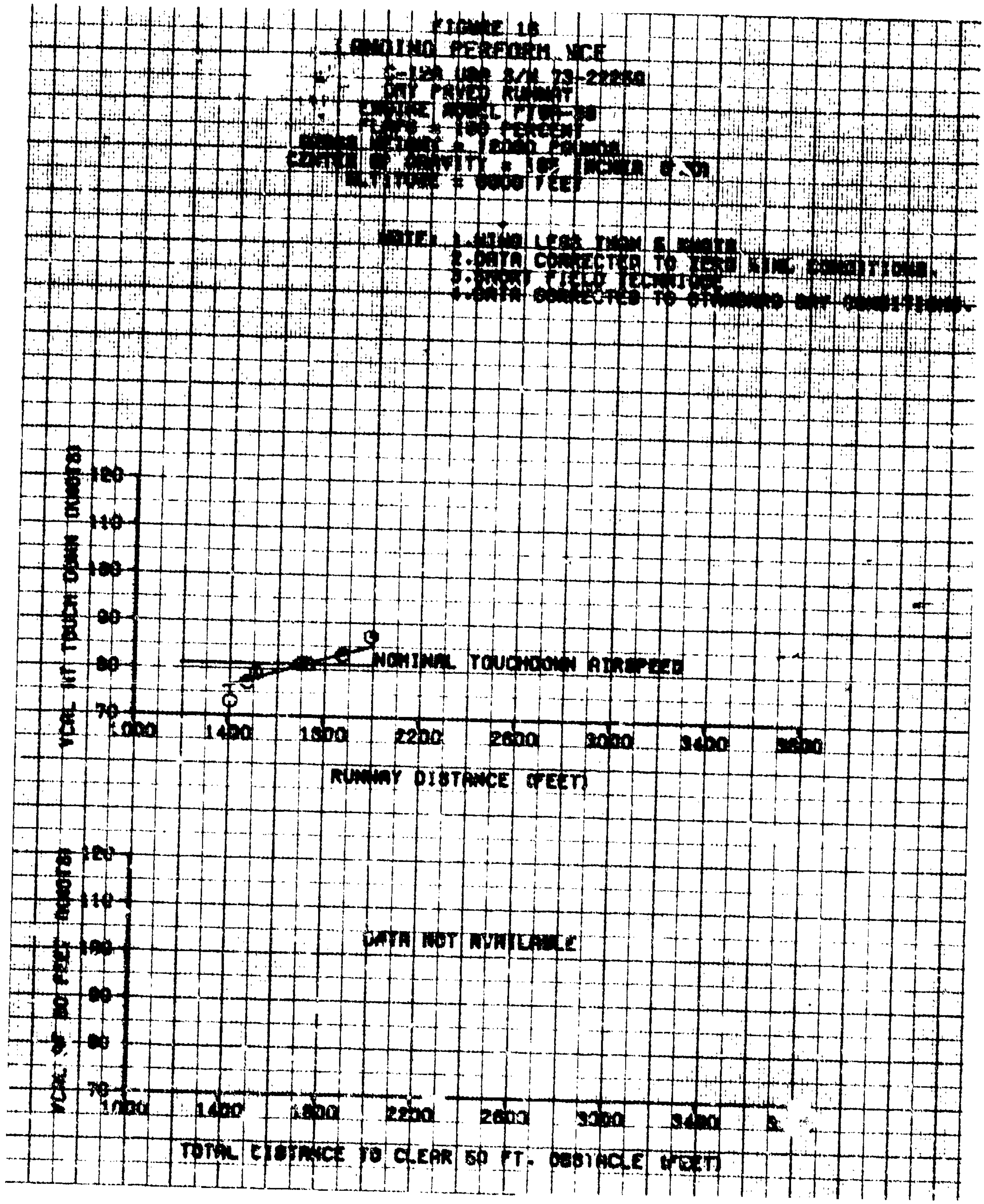


FIGURE 18
LANDING PERFORMANCE
DRY PAVED RUNWAY
2.DATA CORRECTED TO
120
110
100
90
80
70
1000
1400
1800
2200
2600
3000
3400
3800
NOMINAL TOUCHDOWN AIRSPEED
RUNWAY DISTANCE (FEET)
DATA NOT AVAILABLE
TOTAL DISTANCE TO CLEAR 50 FT. OBSTACLE (FEET)

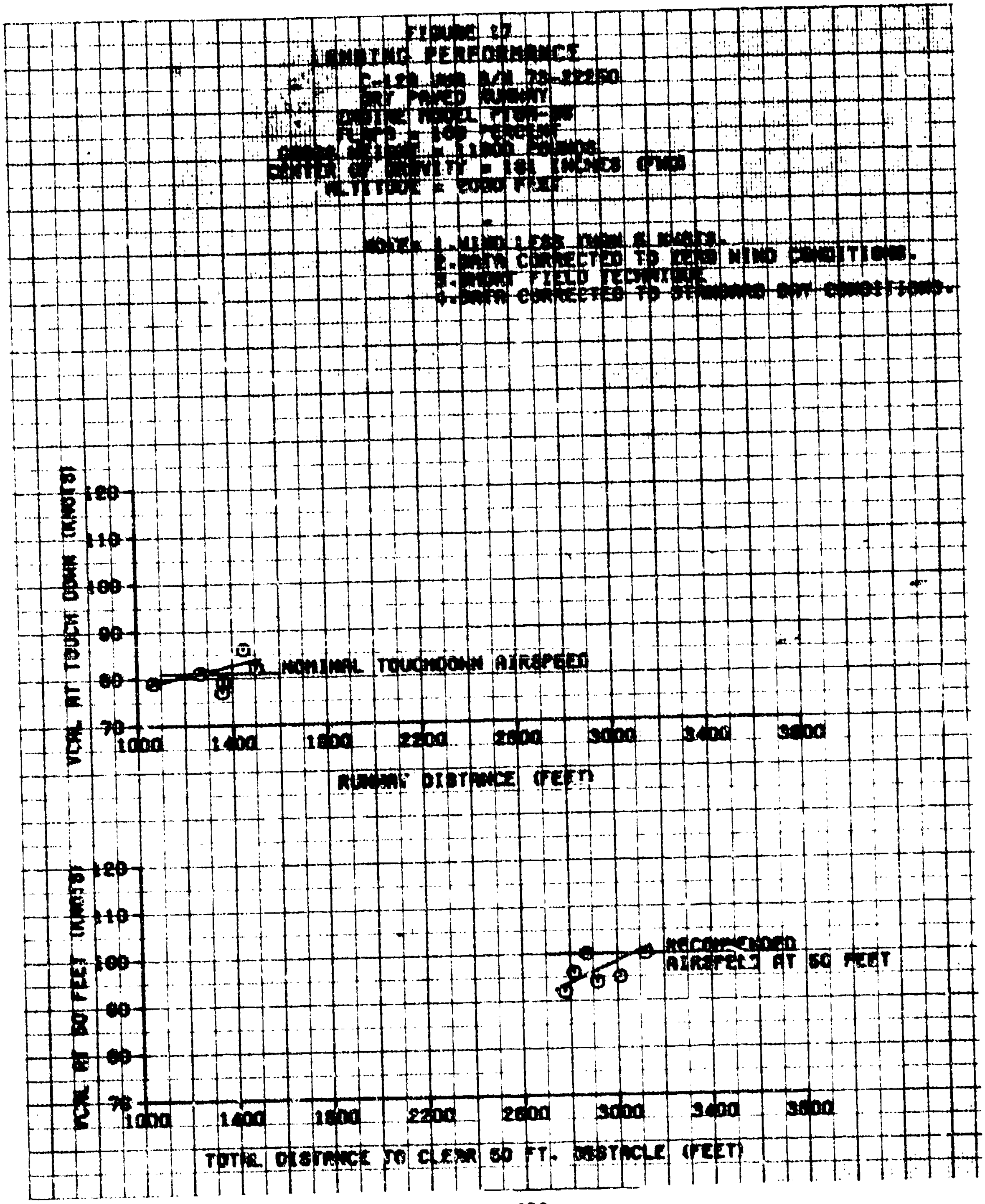


FIGURE 27
LANDING PERFORMANCE
DRY PAVED RUNWAY
2. DATA CORRECTED TO ZERO WIND CONDITIONS.
3. SHORT FIELD TECHNIQUE
4. DATA CORRECTED TO STANDARD DAY CONDITIONS.
NOMINAL TOUCHDOWN AIRSPEED
VCAL AT TOUCH DOWN (KNOTS)
70
80
90
100
110
120
1000
1400
1800
2200
2600
3000
3400
3800
RUNWAY DISTANCE (FEET)
RECOMMENDED
AIRSPEED AT 50 FEET
VCAL AT 50 FEET (KNOTS)
TOTAL DISTANCE TO CLEAR 50 FT. OBSTACLE (FEET)

FIGURE 18

LANDING PERFORMANCE

C-12A USA S/N 73-22250

DRY PAVED RUNWAY

ENGINE MODEL PT6A-38

FLAPS = 100 PERCENT

GROSS WEIGHT = 11000 POUNDS

CENTER OF GRAVITY = 181 INCHES (FWD)

ALTITUDE = 8000 FEET

NOTE: 1. WIND LESS THAN 5 KNOTS.
2. DATA CORRECTED TO ZERO WIND CONDITIONS.
3. SHORT FIELD TECHNIQUE
4. DATA CORRECTED TO STANDARD DAY CONDITIONS.

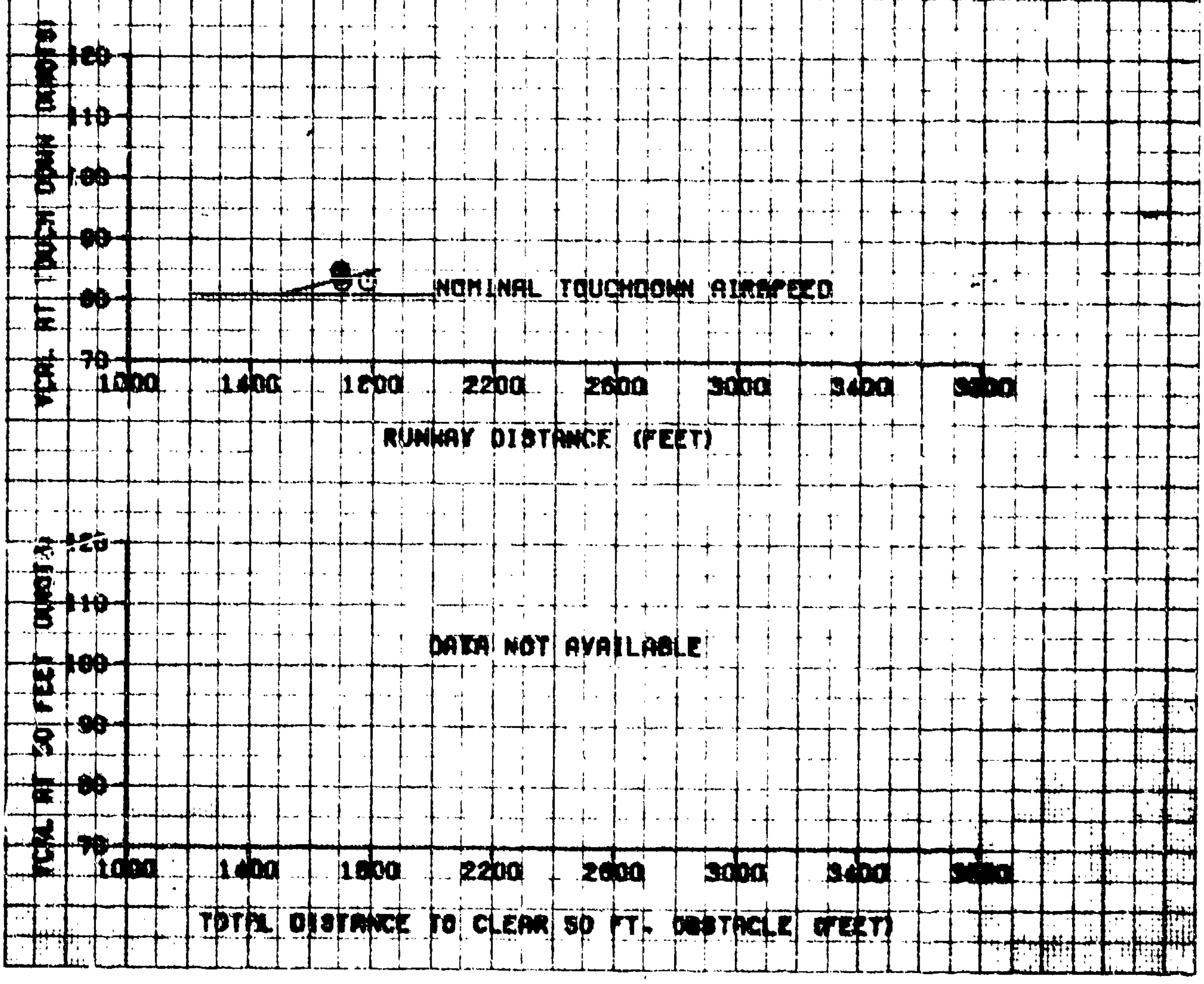


FIGURE 19
LANDING PERFORMANCE
C-12A USA S/N 73-22250
ENGINE MODEL PT6A-38

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	PROPELLER SPEED (RPM)	CONFIGURATION	FLIGHT CONDITION
12390	186.3(FWD)	480	1[illegible]	2000	WAVE OFF	NORMAL LANDING

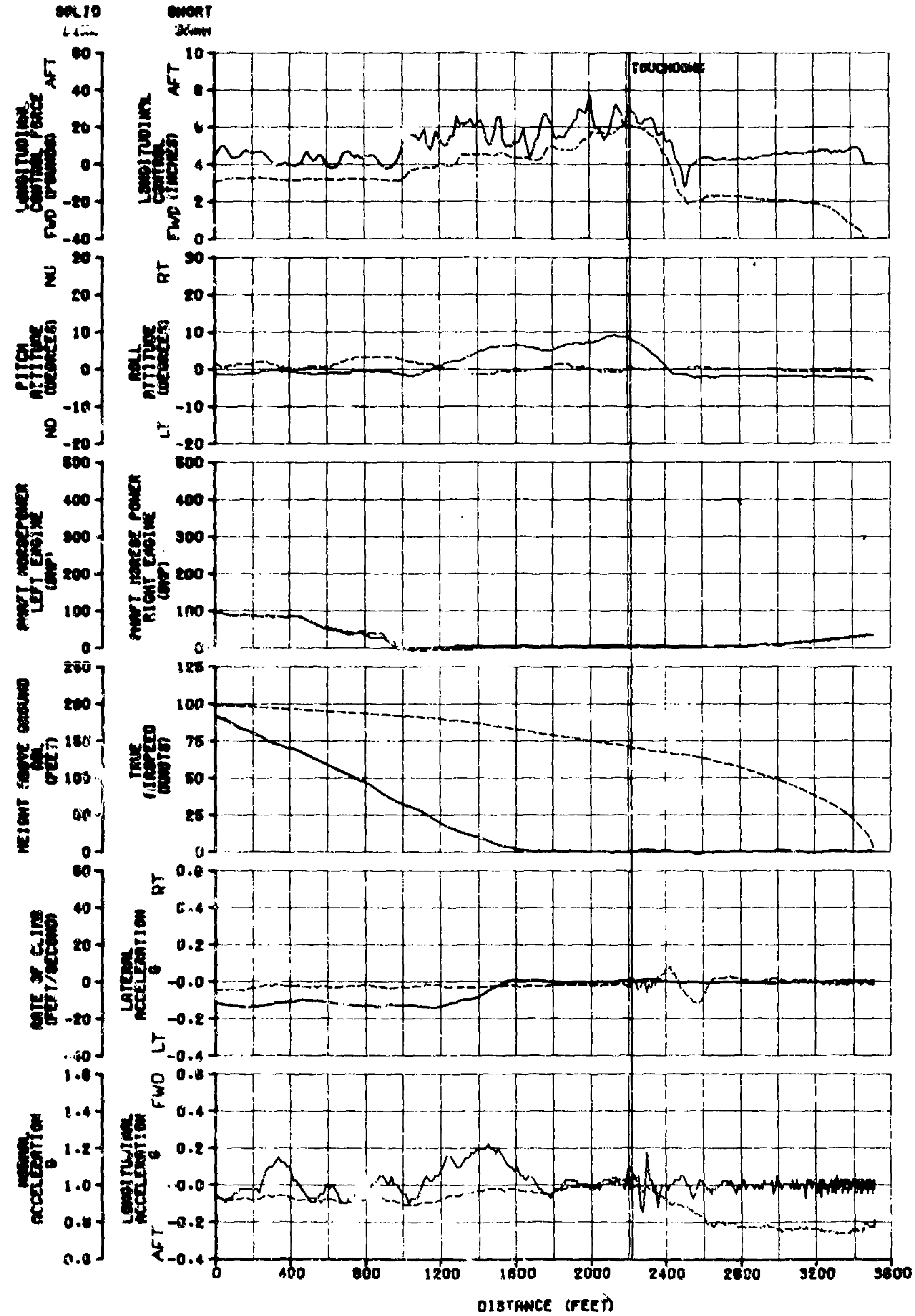
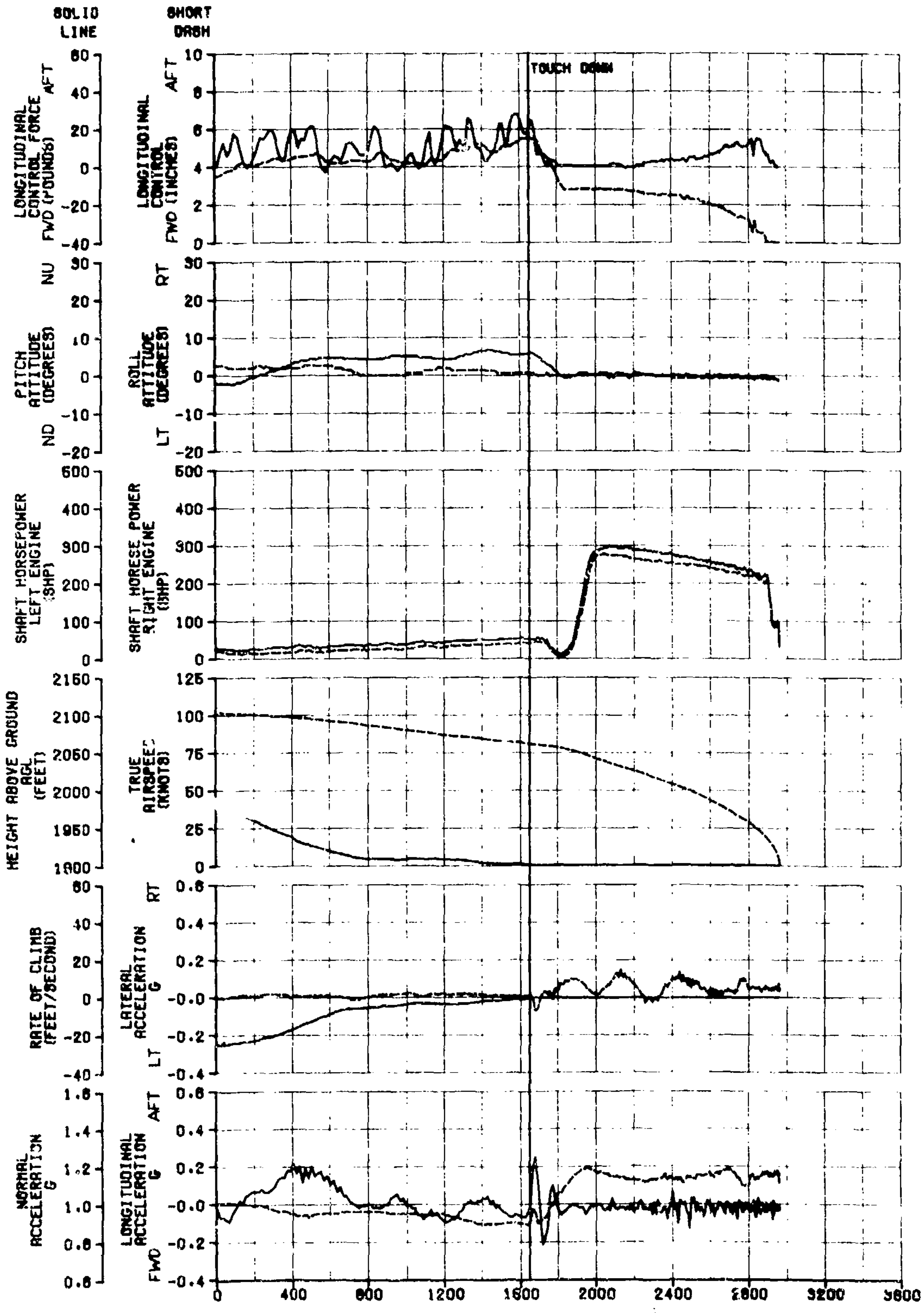
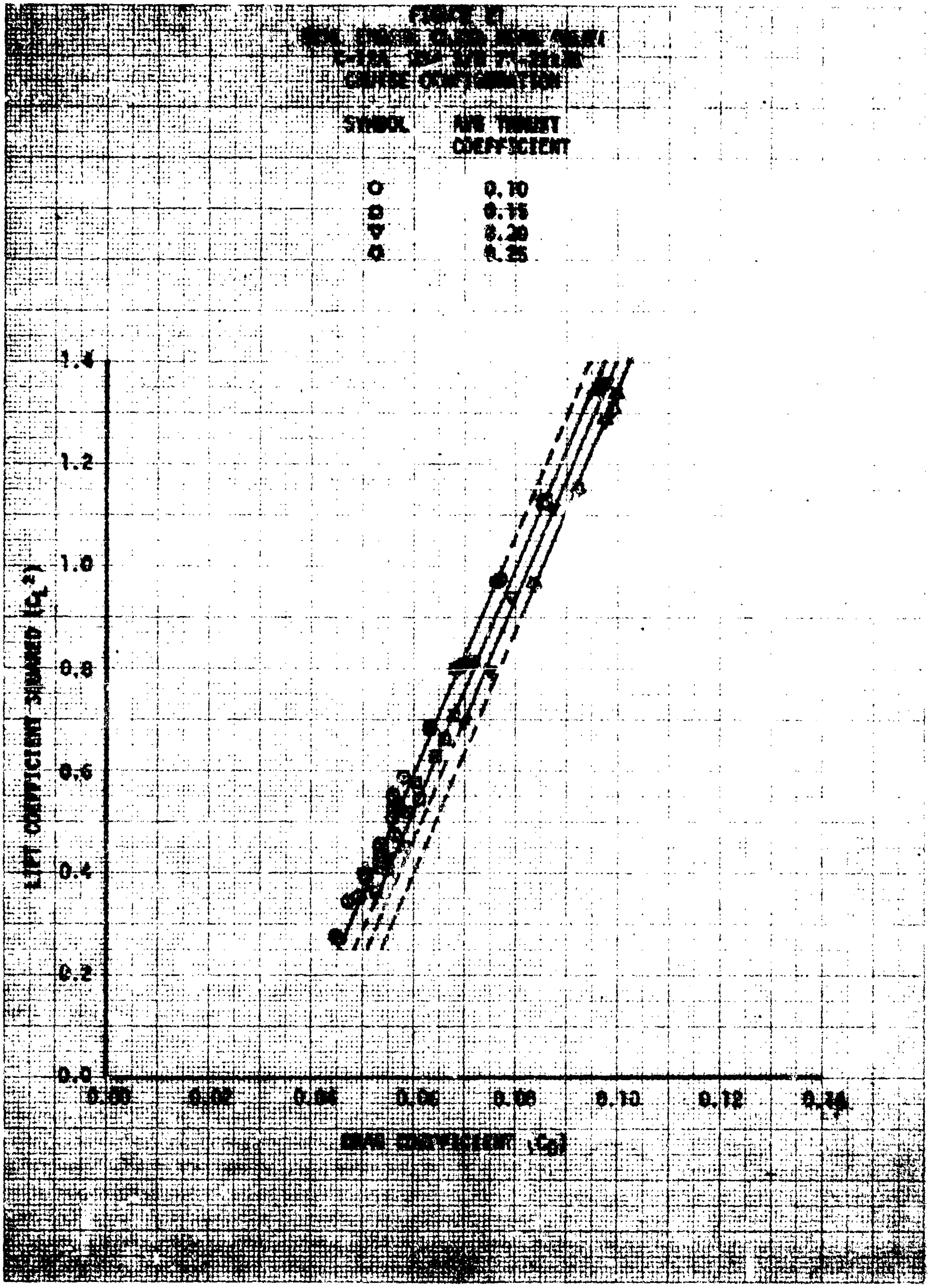


FIGURE 20
LANDING PERFORMANCE
C-12A USA S/N 73-22250
ENGINE MODEL PT6A-38

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	PROPELLER SPEED (RPM)	CONFIGURATION	FLIGHT CONDITION
11340	180.7(FWD)	2130	11.1	2000	WAVE OFF	SHORT FIELD LANDING




SYMBOL
COEFFICIENT
0.10
0.15
0.20
0.25
1.4
1.2
1.0
0.8
0.6
0.4
0.2
0.0
0.04
0.06
0.08
0.10
0.12
0.14

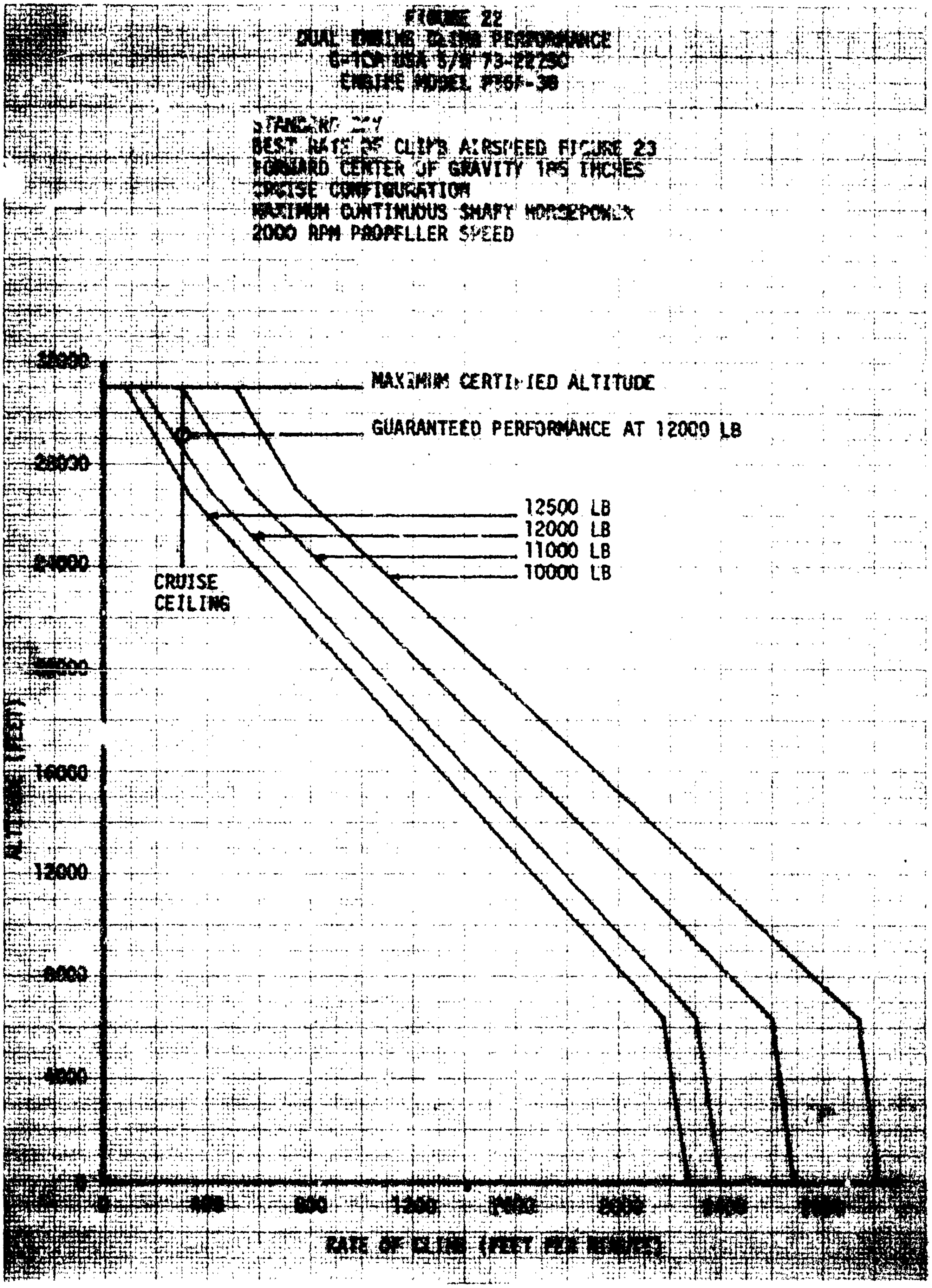


FIGURE 22
DUAL ENGINE CLIMB PERFORMANCE
ENGINE MODEL
STANDARD DAY
BEST RATE OF CLIMB AIRSPEED FIGURE 23
FORWARD CENTER OF GRAVITY INCHES
CRUISE CONFIGURATION
MAXIMUM CONTINUOUS SHAFT HORSEPOWER
2000 RPM PROPELLER SPEED
MAXIMUM CERTIFIED ALTITUDE
GUARANTEED PERFORMANCE AT 12000 LB
12500 LB
12000 LB
11000 LB
10000 LB
CRUISE
CEILING
28000
24000
16000
12000
8000
4000
0
0
400
800
1200
1600
2000
2400
2800
RATE OF CLIMB (FEET PER MINUTE)

FIGURE 23
DUAL ENGINE CLIMB PERFORMANCE
C-12A USA S/N 78-23290
ENGINE MODEL PT6A-38

STANDARD DAY
FORWARD CENTER OF GRAVITY 185 INCHES
CRUISE CONFIGURATION
MAXIMUM CONTINUOUS SHAFT HORSEPOWER
2000 RPM PROPELLER SPEED

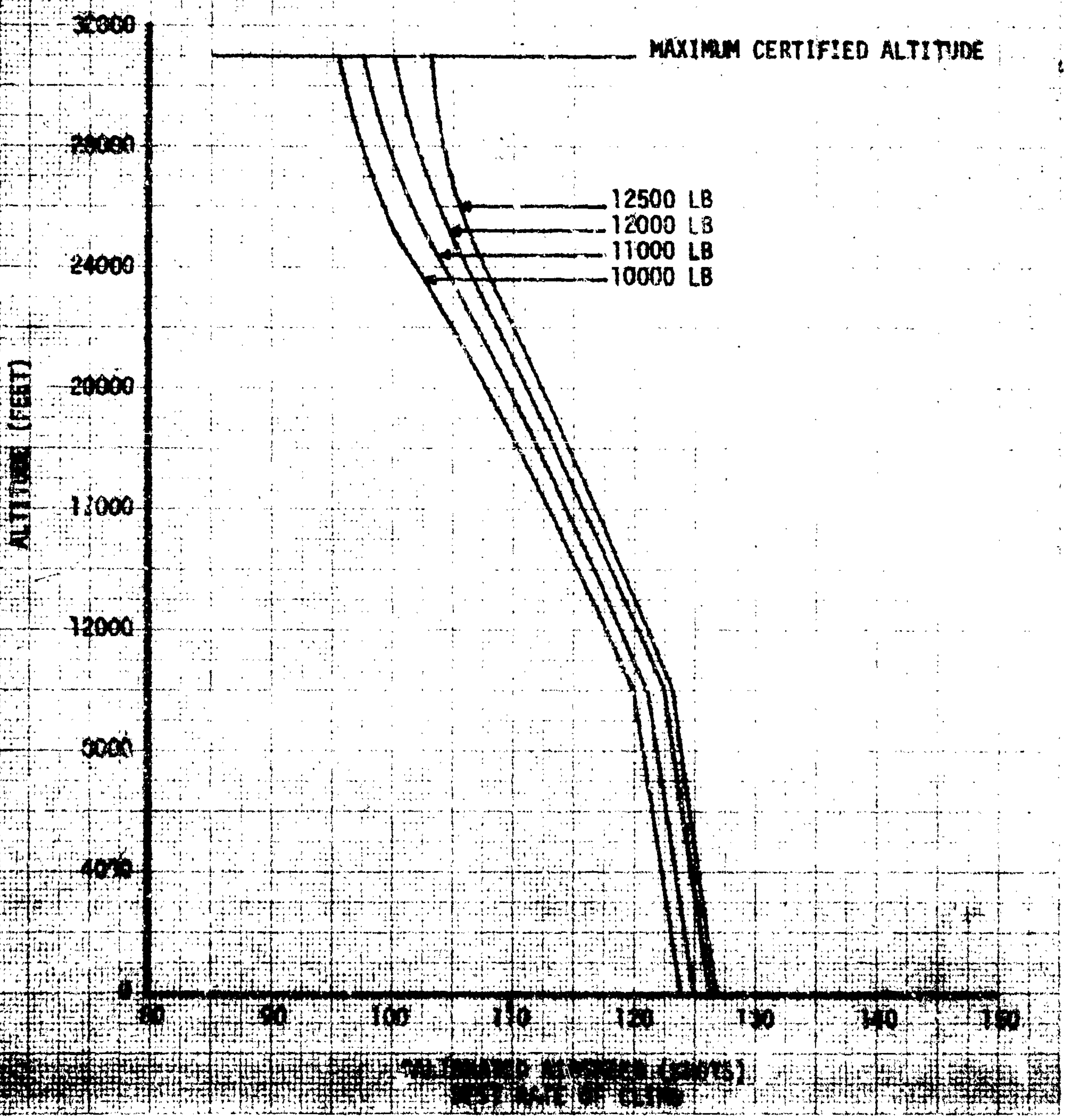
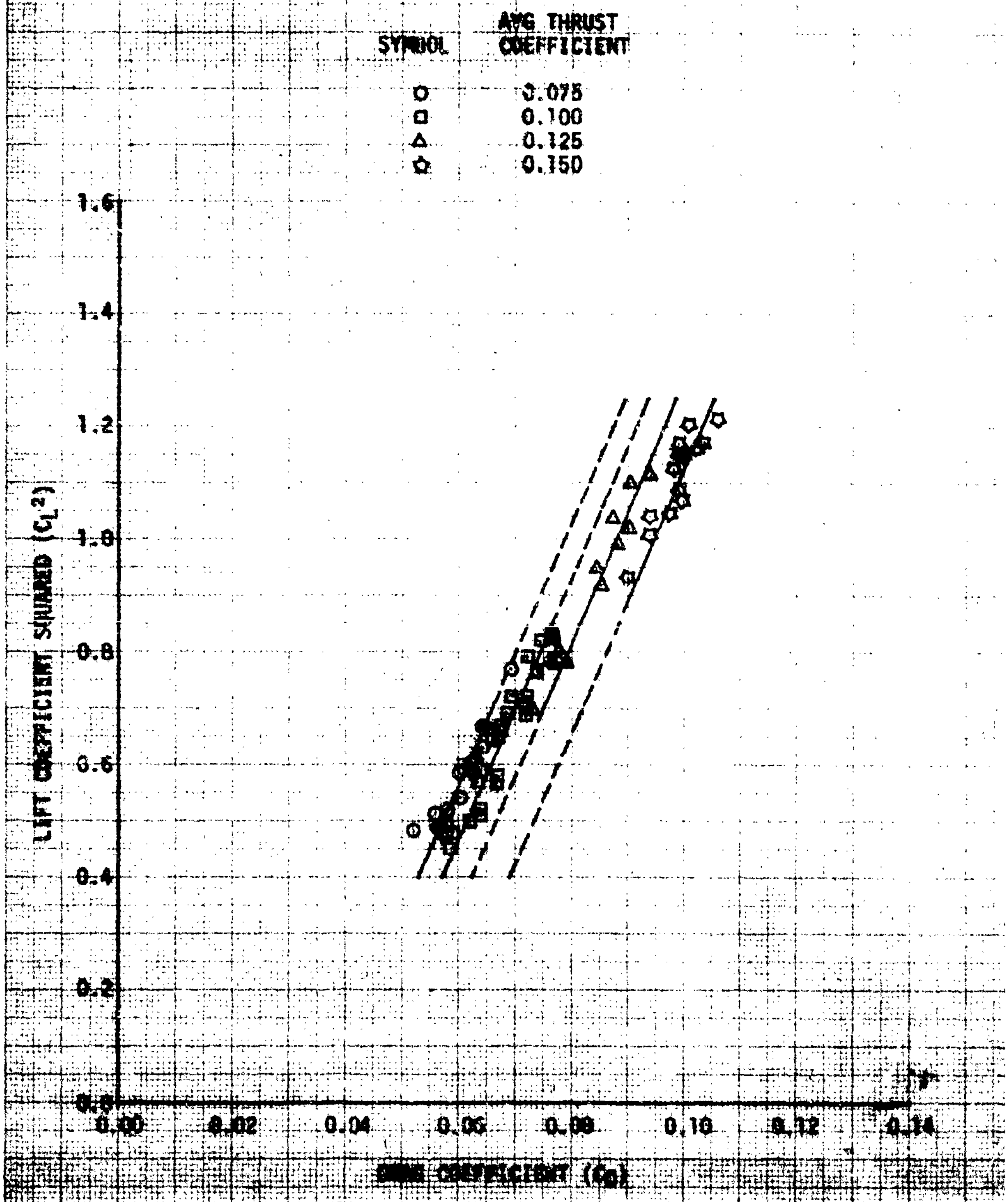
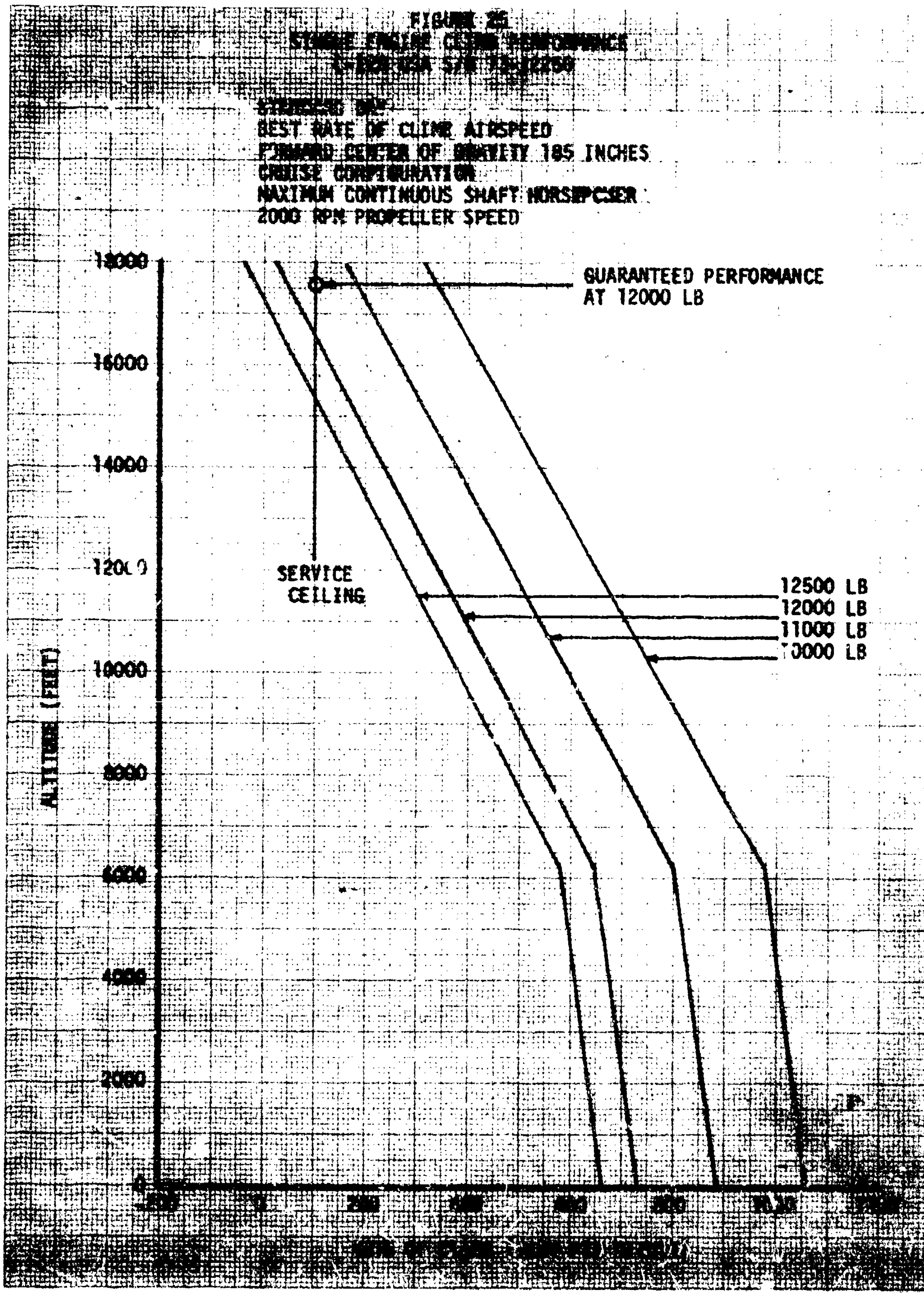


FIGURE 24
SINGLE ENGINE CLIMB DRAG POLAR
C-12A USA S/N 73-22290
CRUISE CONFIGURATION
LEFT ENGINE INOPERATIVE AND PROPELLOR FEATHERED





BEST RATE OF CLIMB AIRSPEED
185 INCHES
MAXIMUM CONTINUOUS SHAFT
2000 RPM PROPELLER SPEED
GUARANTEED PERFORMANCE
AT 12000 LB
SERVICE
CEILING
12500 LB
12000 LB
11000 LB
0000 LB
18000
16000
14000
10000
8000
6000
4000
2000
0
200

FIGURE 26
SINGLE ENGINE CLIMB PERFORMANCE
C-12A USA S/N 73-22250
ENGINE MODEL PT6A-38

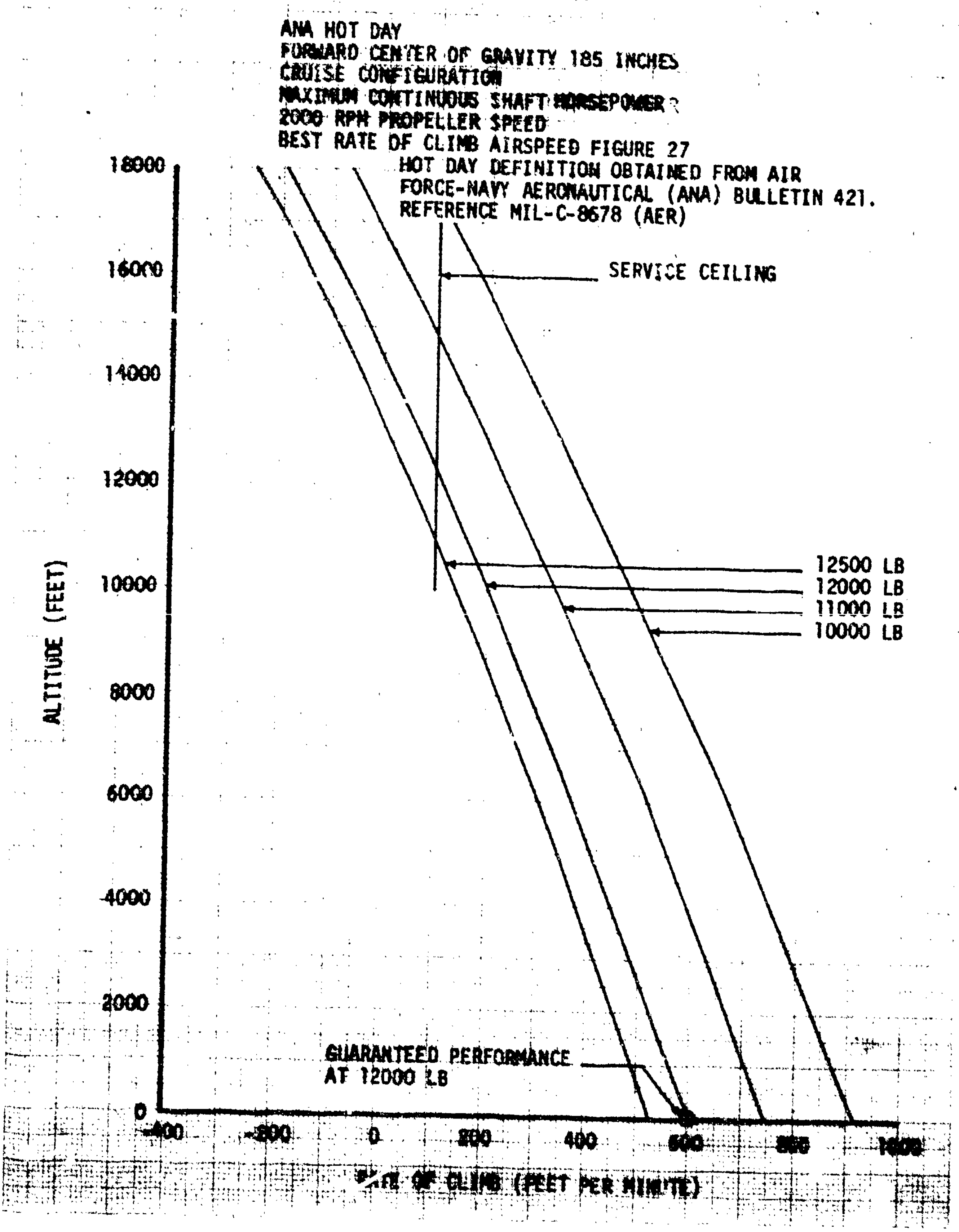


FIGURE 27
SINGLE ENGINE CLIMB PERFORMANCE
C-12A USA S/N 73-22250
ENGINE MODEL PT6A-38

STANDARD DAY
FORWARD CENTER OF GRAVITY 185 INCHES
CRUISE CONFIGURATION
MAXIMUM CONTINUOUS SHAFT HORSEPOWER
2000 RPM PROPELLER SPEED

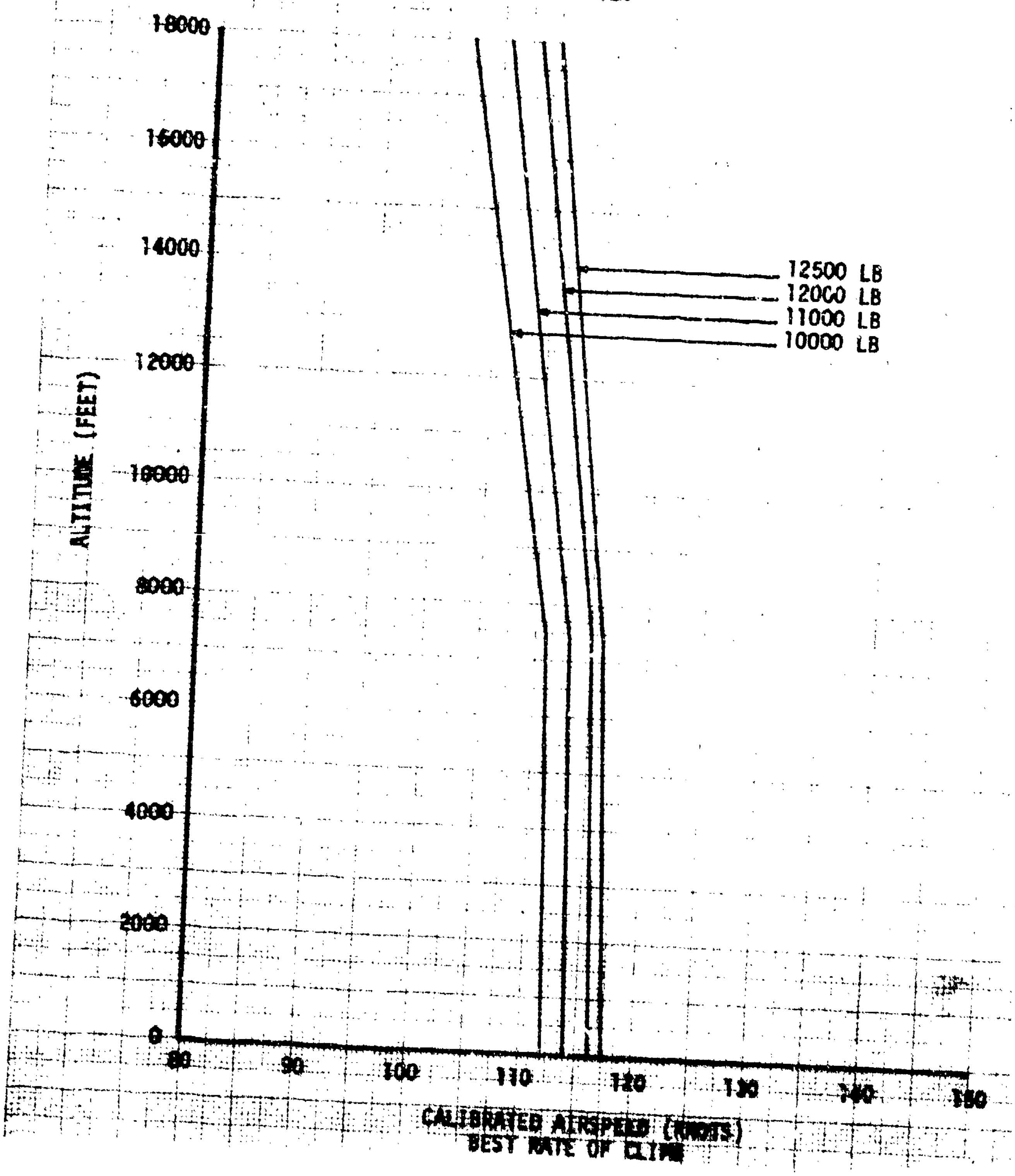
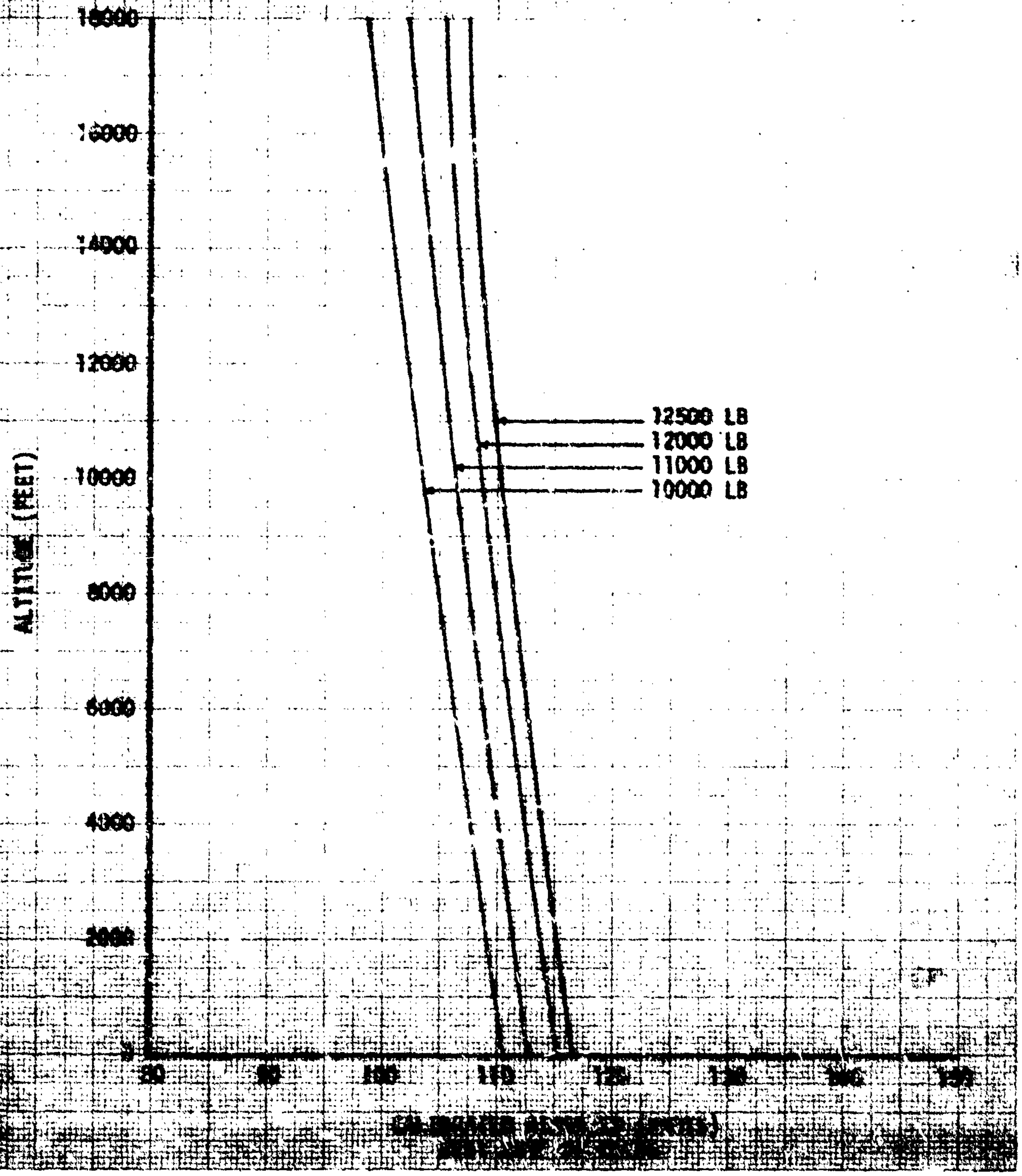
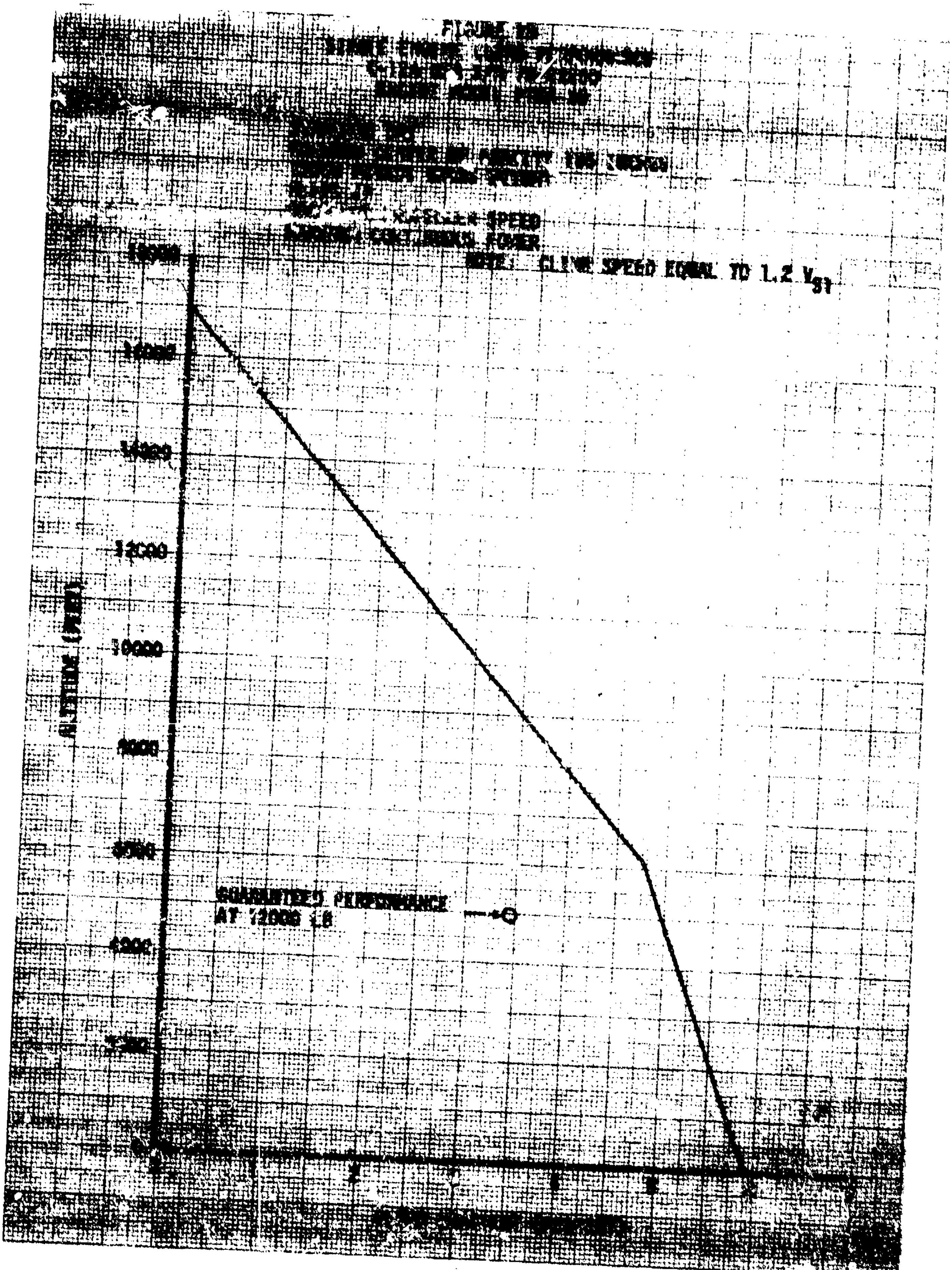


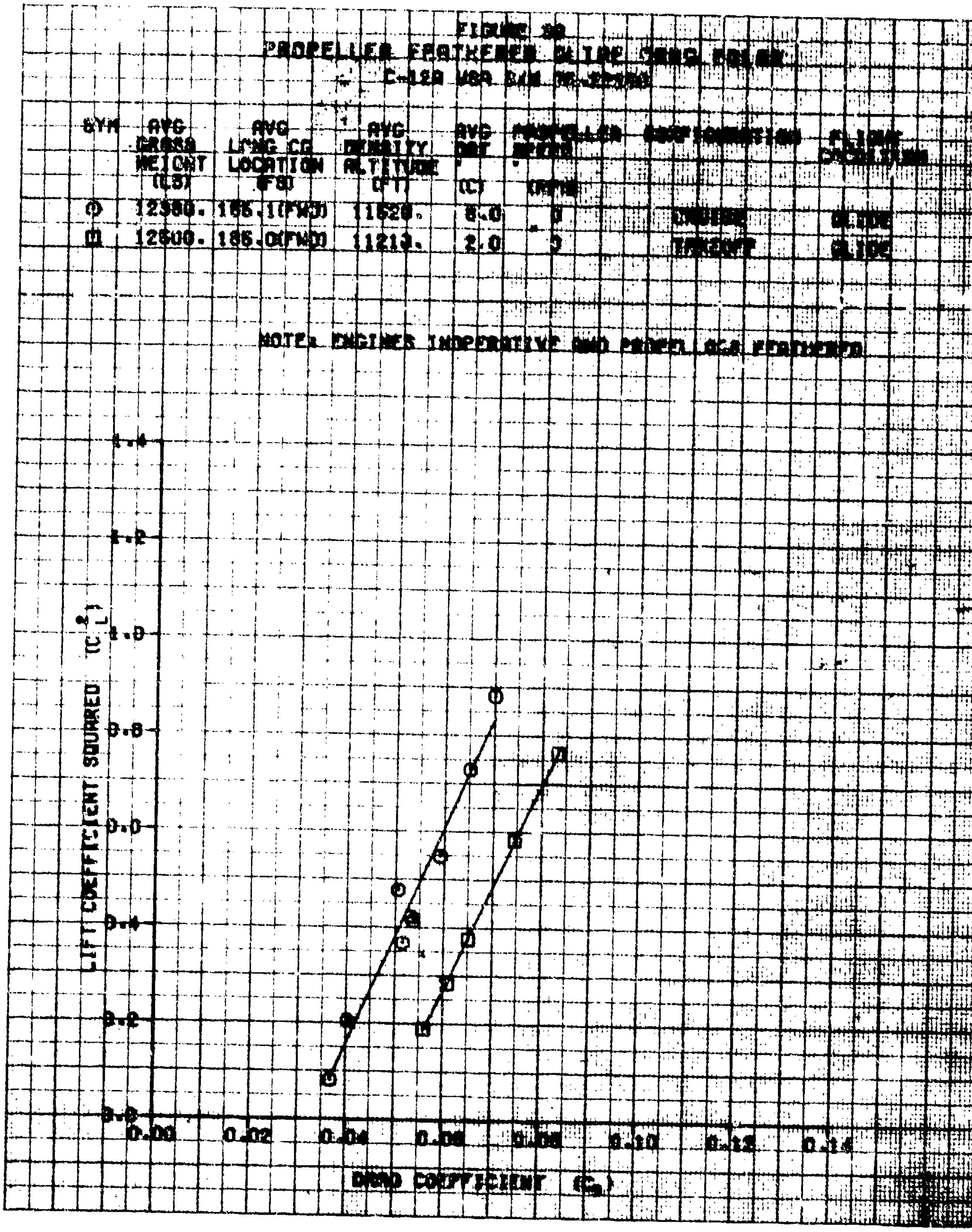
FIGURE [illegible]
SINGLE ENGINE [illegible] PERFORMANCE
C-2[illegible]A USA S/N 7[illegible]-22[illegible]
ENGINE MODEL P[illegible]6A-38

[illegible] HOT DAY
FORWARD CENTER OF GRAVITY 1[illegible] INCHES
[illegible] CONFIGU[illegible]
MAXIMUM CONTINUOUS SHAFT HORSEPOWER
2000 RPM PROPELLER SPEED

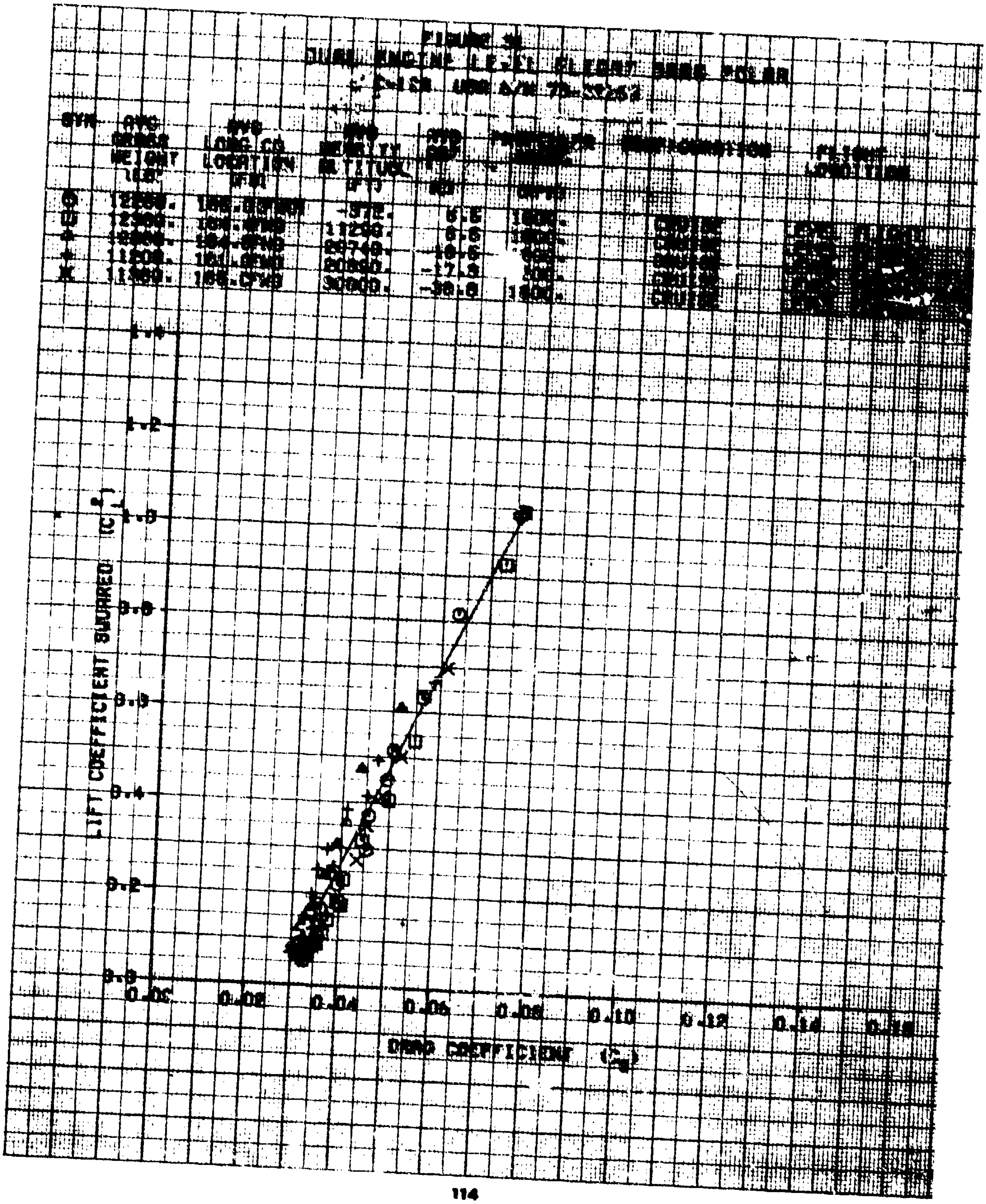




SPEED
CONTINUOUS POWER
NOTE: CLIMB SPEED EQUAL TO 1.2 V_{S1}
16000
14000
12000
10000
8000
6000
4000
ALTITUDE (FEET)
GUARANTEED PERFORMANCE
AT 12000 LB



SYM
AVG GROSS WEIGHT (LB)
AVG LOCATION
AVG ALTITUDE (FT)
AVG (C)
CRUISE
TAKEOFF
GLIDE
GLIDE
LIFT COEFFICIENT SQUARED
0.00
0.02
0.04
0.06
0.08
0.10
0.12
0.14
DRAG COEFFICIENT


LIFT COEFFICIENT SQUARED
1.2
1.0
0.4
0.2
0.04
0.06
0.08
0.10
0.12
0.14
0.16
DRAG COEFFICIENT

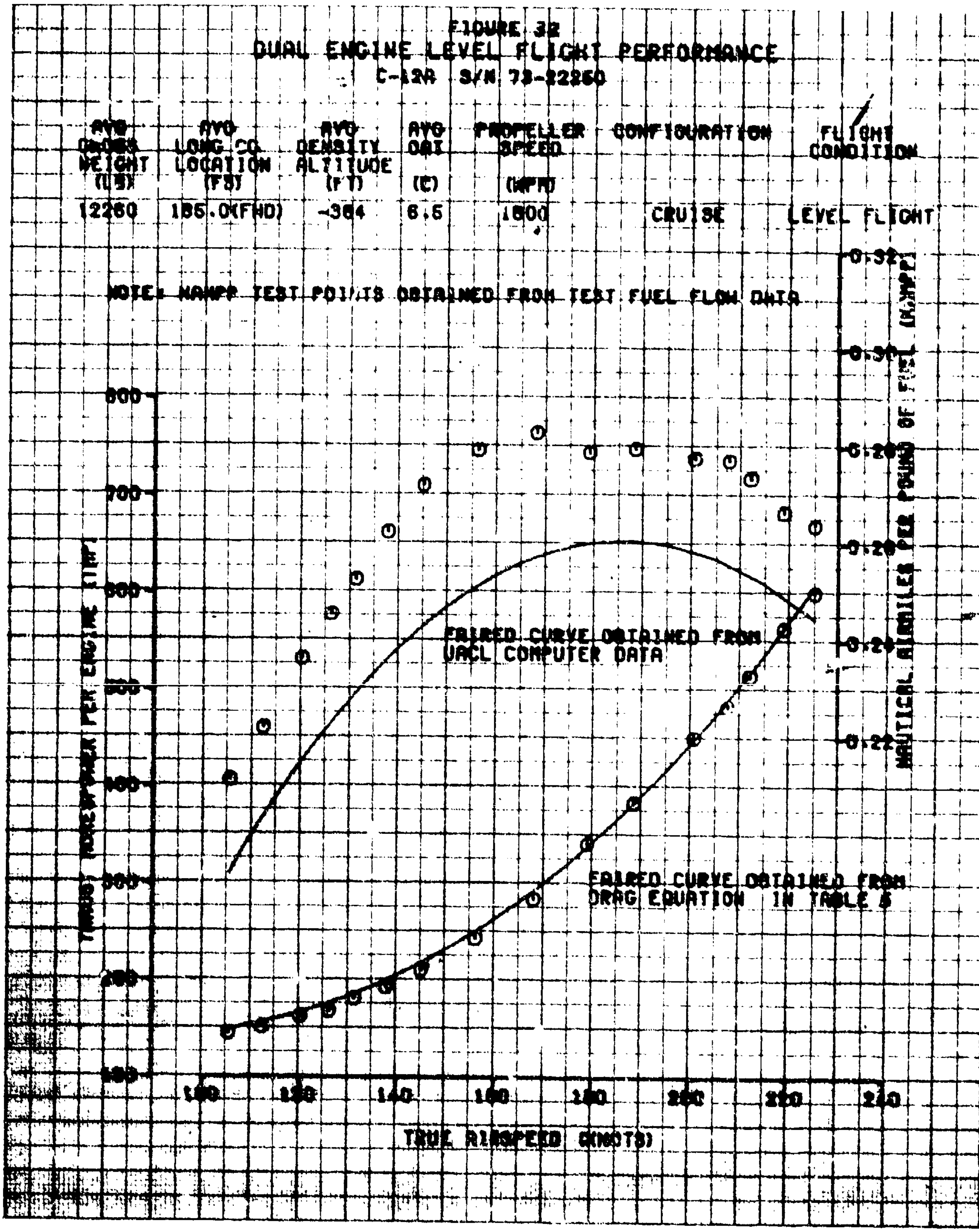
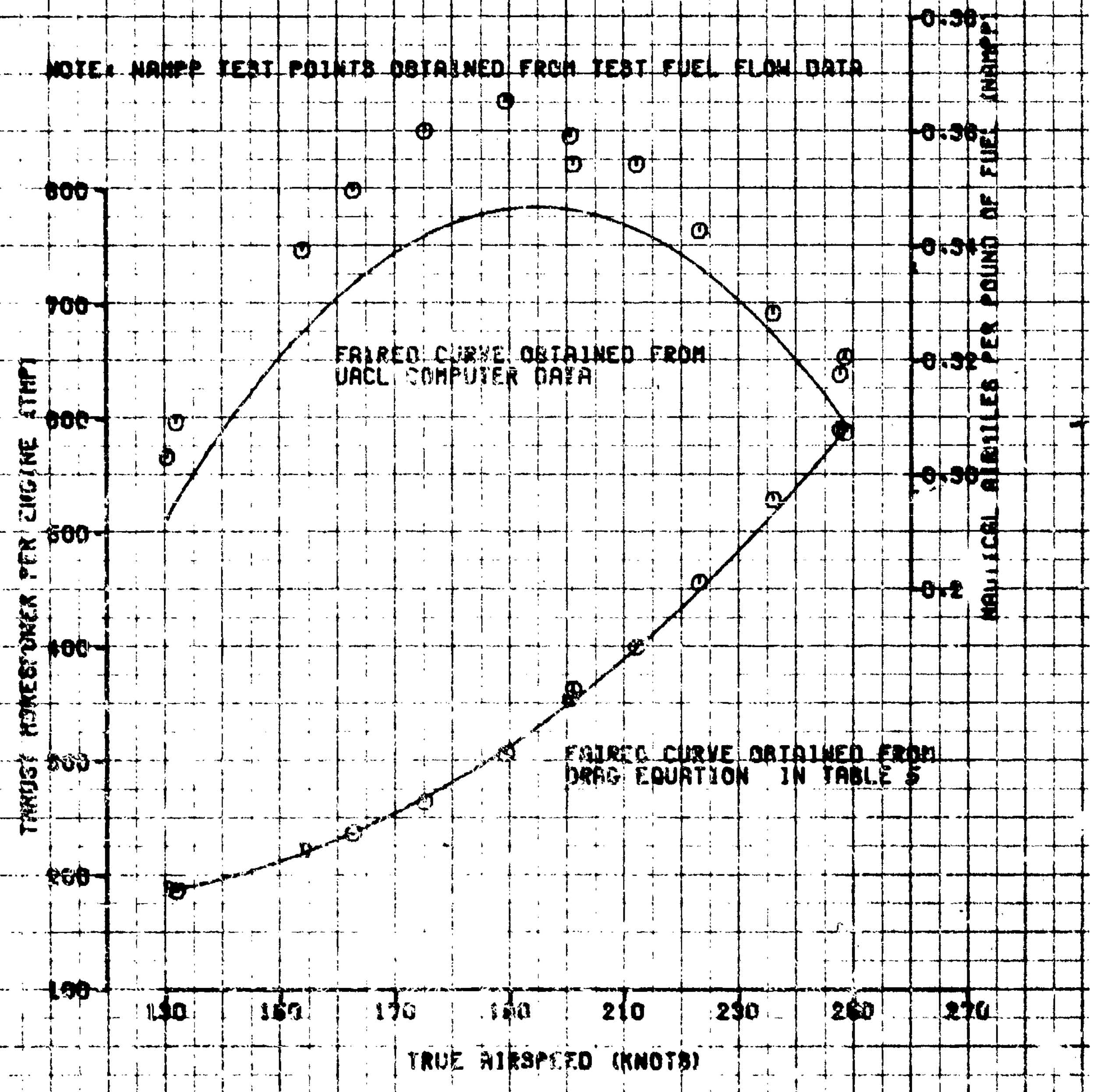


FIGURE 32
DUAL ENGINE LEVEL FLIGHT PERFORMANCE
C-12A S/N 73-22250
AVG GROSS WEIGHT (LB)
AVG LONG CG LOCATION (FS)
AVG DENSITY ALTITUDE (FT)
AVG OAT (C)
PROPELLER SPEED (RPM)
CONFIGURATION
FLIGHT CONDITION
12250
185.0(FWD)
-384
6.5
1800
CRUISE
LEVEL FLIGHT
NOTE: NAMPP TEST POINTS OBTAINED FROM TEST FUEL FLOW DATA
FAIRED CURVE OBTAINED FROM UACL COMPUTER DATA
FAIRED CURVE OBTAINED FROM DRAG EQUATION IN TABLE 5
THRUST HORSEPOWER PER ENGINE (THP)
NAUTICAL AIRMILES PER POUND OF FUEL (NAMPP)
TRUE AIRSPEED (KNOTS)
100
200
300
400
500
600
700
800
0.22
0.24
0.26
0.28
0.30
0.32
100
120
140
160
180
200
220
240

FIGURE 39
DUAL ENGINE LEVEL FLIGHT PERFORMANCE
C-12A S/N 73-22250

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	PROPELLER SPEED (RPM)	CONFIGURATION	FLIGHT CONDITION
12440	194.8(FWD)	11300	6.6	1800	CRUISE	LEVEL FLIGHT



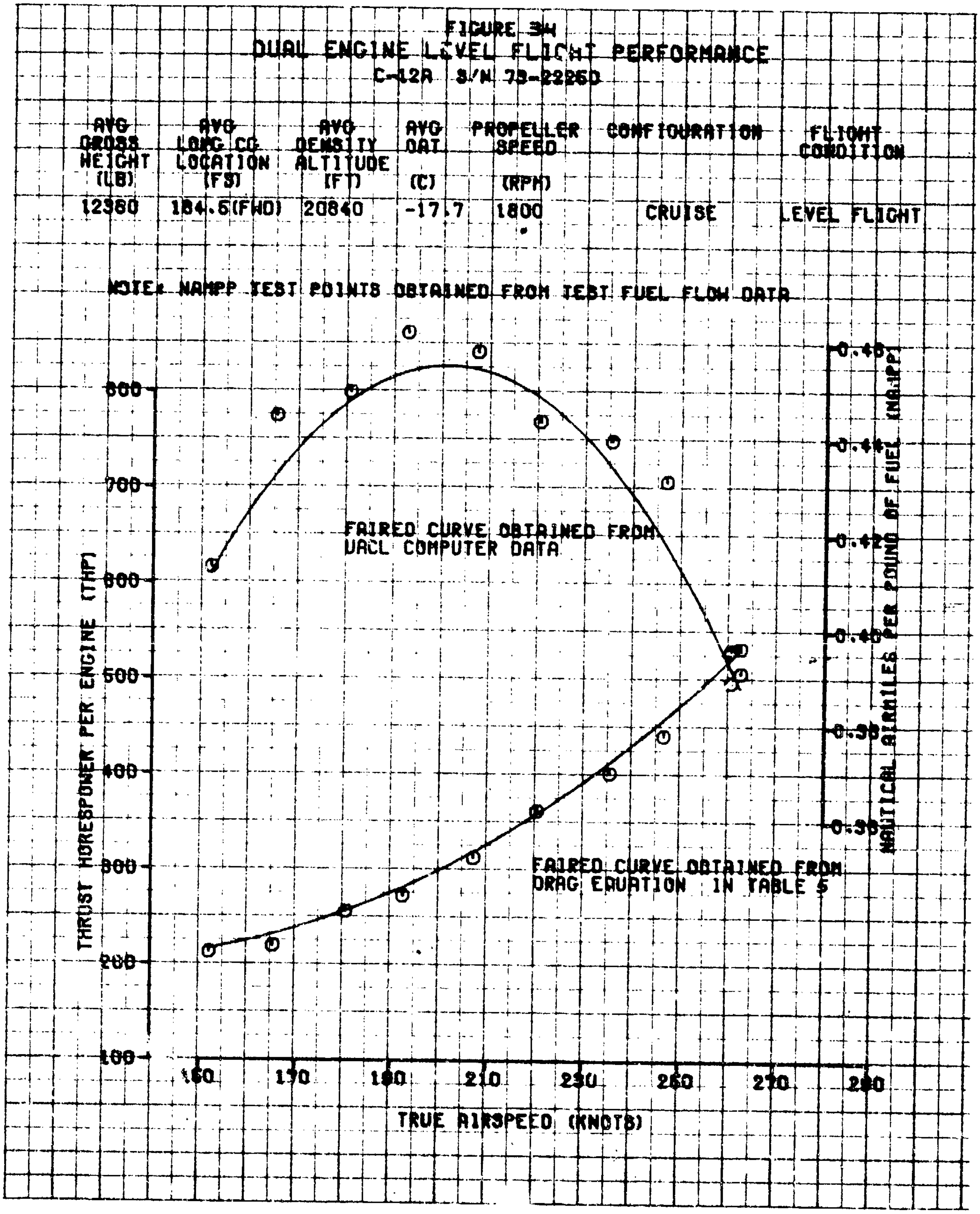


FIGURE 34
DUAL ENGINE LEVEL FLIGHT PERFORMANCE
C-12A S/N 73-22250
AVG GROSS WEIGHT (LB) 12380
AVG LONG CG LOCATION (FS) 184.5(FWD)
AVG DENSITY ALTITUDE (FT) 20840
AVG OAT (C) -17.7
PROPELLER SPEED (RPM) 1800
CONFIGURATION CRUISE
FLIGHT CONDITION LEVEL FLIGHT
NOTE: NAMPP TEST POINTS OBTAINED FROM TEST FUEL FLOW DATA
FAIRED CURVE OBTAINED FROM UACL COMPUTER DATA
FAIRED CURVE OBTAINED FROM DRAG EQUATION IN TABLE 5
THRUST HORSEPOWER PER ENGINE (THP)
NAUTICAL AIRMILES PER POUND OF FUEL (NAMPP)
TRUE AIRSPEED (KNOTS)
100
200
300
400
500
600
700
800
0.36
0.38
0.40
0.42
0.44
0.46
160
170
180
210
230
250
270
290

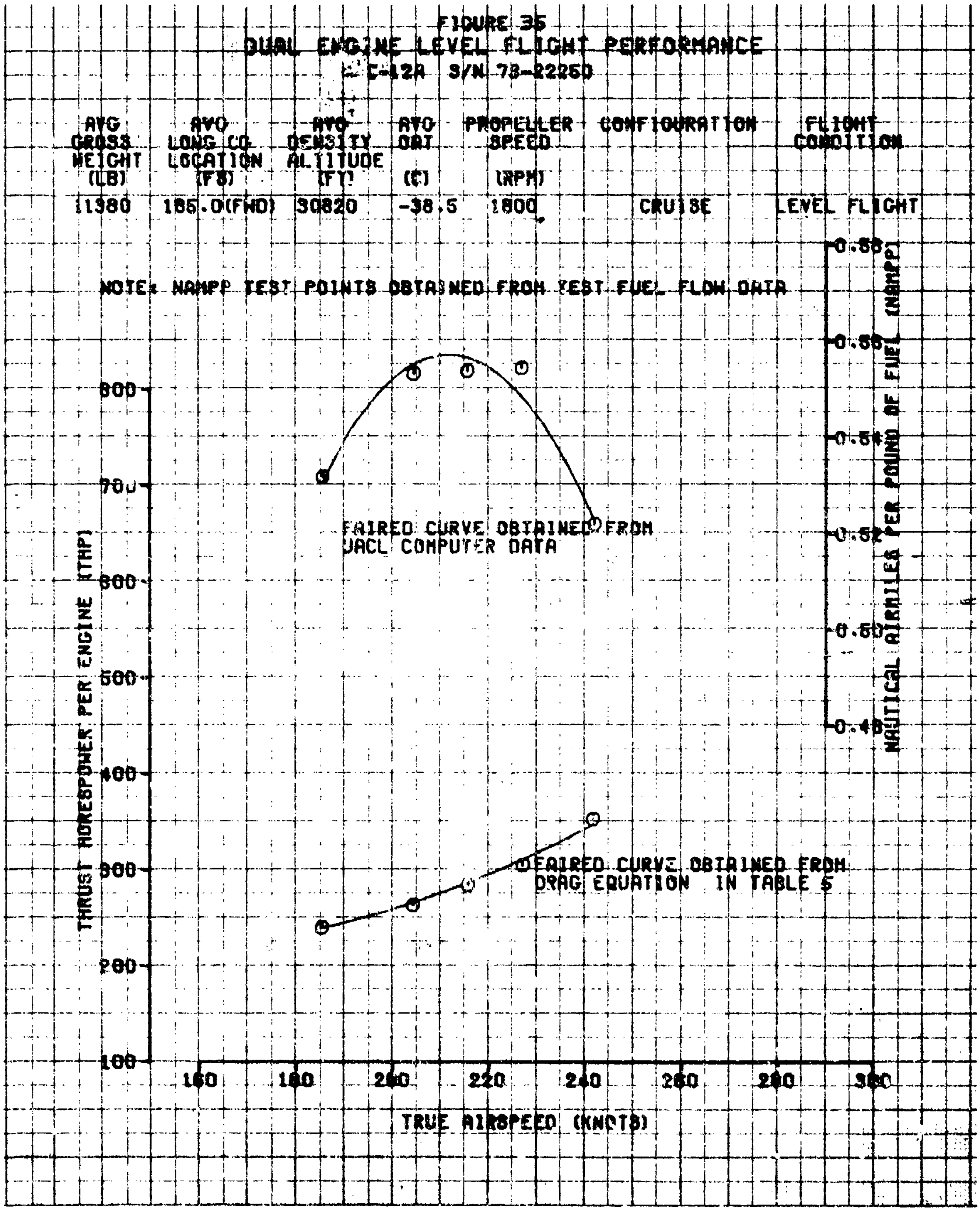


FIGURE 35
DUAL ENGINE LEVEL FLIGHT PERFORMANCE
C-12A S/N 73-22250
AVG GROSS WEIGHT (LB)
AVG LONG CG LOCATION (FS)
AVG DENSITY ALTITUDE (FT)
AVG OAT (C)
PROPELLER SPEED (RPM)
CONFIGURATION
FLIGHT CONDITION
11380
185.0(FWD)
30820
-38.5
1800
CRUISE
LEVEL FLIGHT
NOTE: NAMPP TEST POINTS OBTAINED FROM TEST FUEL FLOW DATA
FAIRED CURVE OBTAINED FROM
UACL COMPUTER DATA
FAIRED CURVE OBTAINED FROM
DRAG EQUATION IN TABLE 5
THRUST HORSEPOWER PER ENGINE (THP)
NAUTICAL AIRMILES PER POUND OF FUEL (NAMPP)
TRUE AIRSPEED (KNOTS)

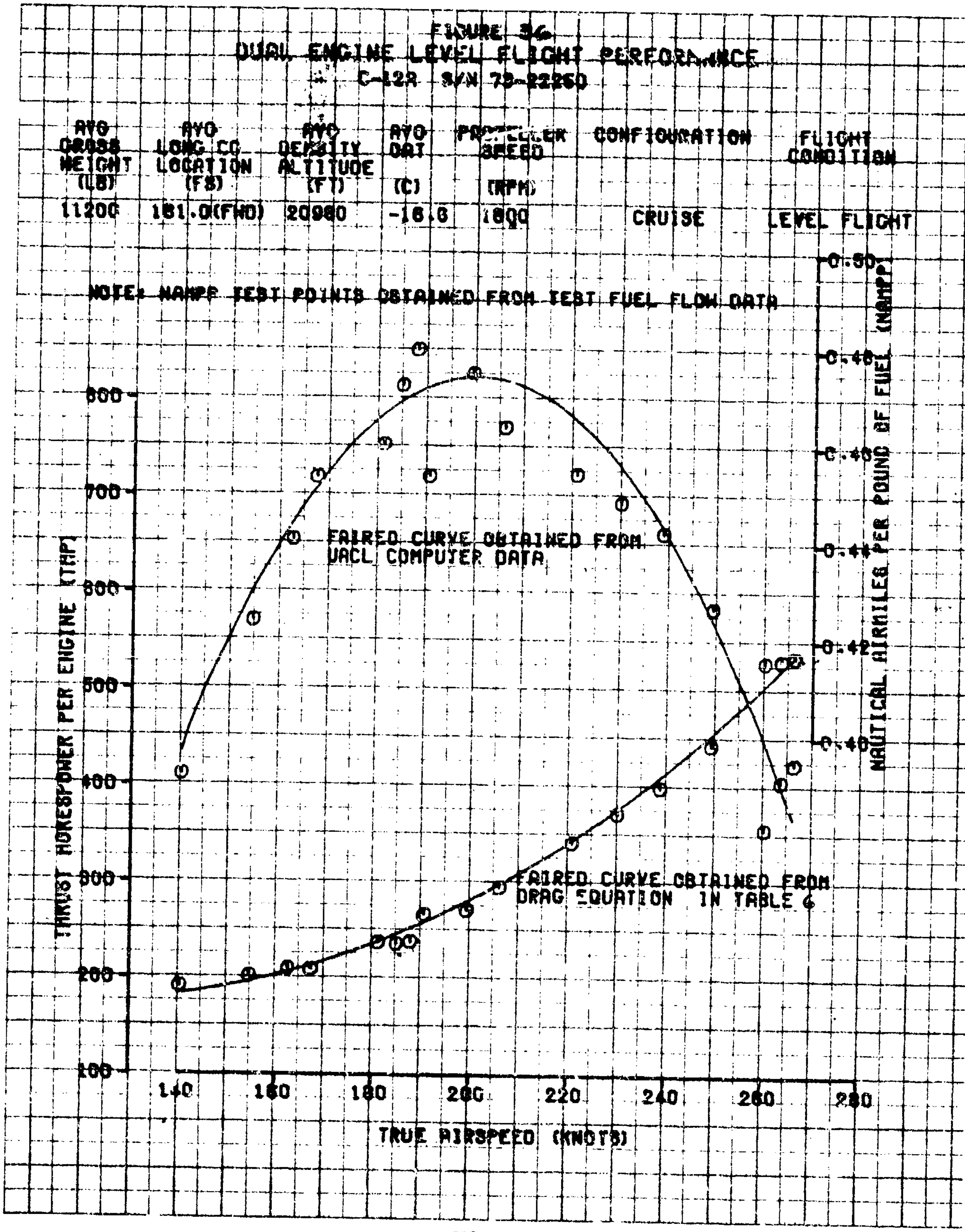


FIGURE 36
DUAL ENGINE LEVEL FLIGHT PERFORMANCE
C-12A S/N 73-22250
AVG GROSS WEIGHT (LB)
AVG LONG CG LOCATION (FS)
AVG DENSITY ALTITUDE (FT)
AVG OAT (C)
PROPELLER SPEED (RPM)
CONFIGURATION
FLIGHT CONDITION
11200
181.0(FWD)
20960
-16.6
1800
CRUISE
LEVEL FLIGHT
NOTE: NAMPP TEST POINTS OBTAINED FROM TEST FUEL FLOW DATA
FAIRED CURVE OBTAINED FROM UACL COMPUTER DATA
FAIRED CURVE OBTAINED FROM DRAG EQUATION IN TABLE 6
THRUST HORSEPOWER PER ENGINE (THP)
NAUTICAL AIRMILES PER POUND OF FUEL (NAMPP)
TRUE AIRSPEED (KNOTS)
100
200
300
400
500
600
700
800
0.40
0.42
0.44
0.46
0.48
0.50
140
160
180
200
220
240
260
280

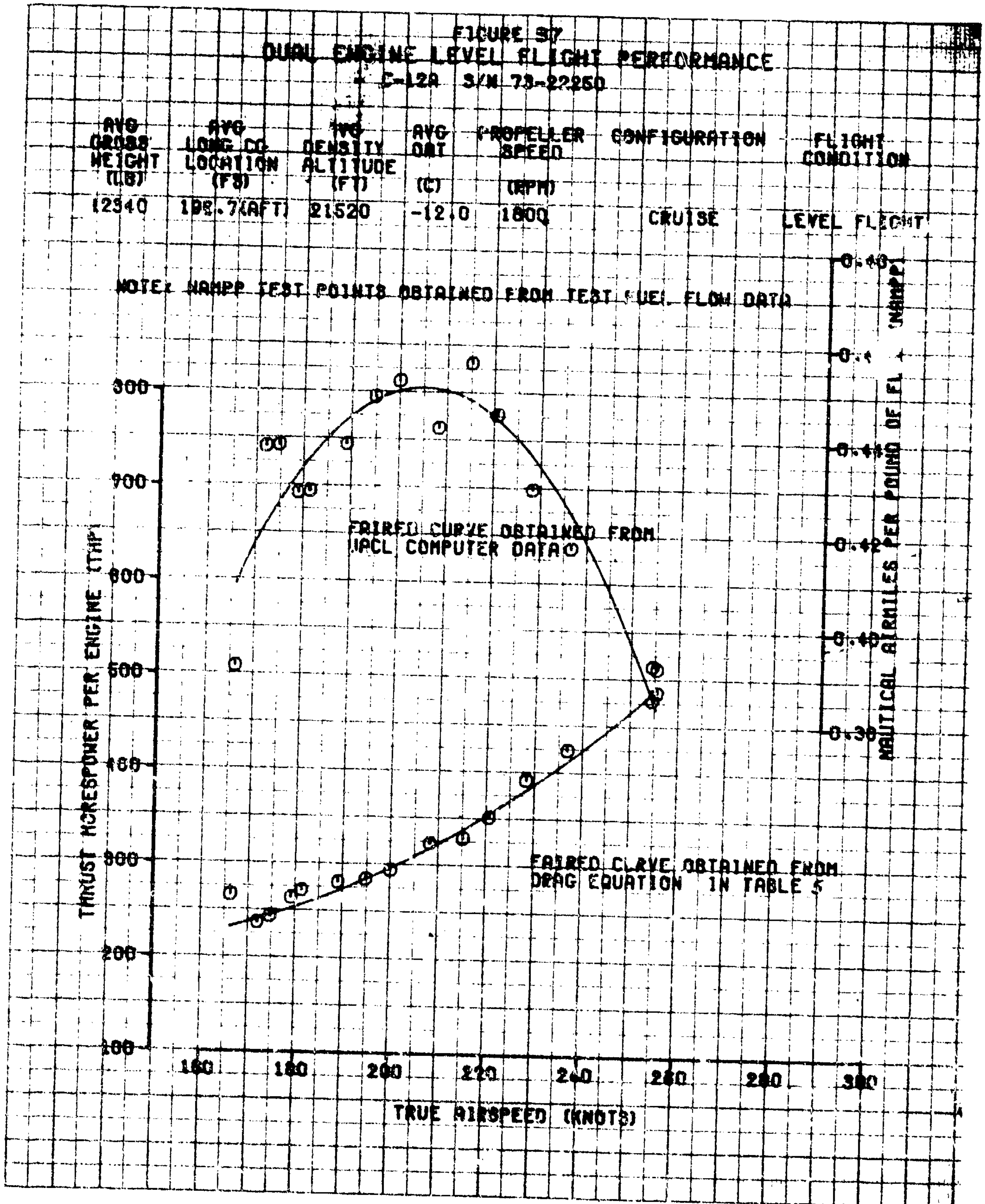


FIGURE 37
DUAL ENGINE LEVEL FLIGHT PERFORMANCE
C-12A S/N 73-22250
AVG GROSS WEIGHT (LB)
AVG LONG CG LOCATION (FS)
AVG DENSITY ALTITUDE (FT)
AVG OAT (C)
PROPELLER SPEED (RPM)
CONFIGURATION
FLIGHT CONDITION
12340
195.7(AFT)
21520
-12.0
1800
CRUISE
LEVEL FLIGHT
NOTE: NAMPP TEST POINTS OBTAINED FROM TEST FUEL FLOW DATA
FAIRED CURVE OBTAINED FROM UACL COMPUTER DATA
FAIRED CURVE OBTAINED FROM DRAG EQUATION IN TABLE 5
THRUST HORSEPOWER PER ENGINE (THP)
NAUTICAL AIRMILES PER POUND OF FUEL (NAMPP)
TRUE AIRSPEED (KNOTS)
300
700
600
500
400
300
200
100
160
180
200
220
240
260
280
300
0.46
0.42
0.40
0.38

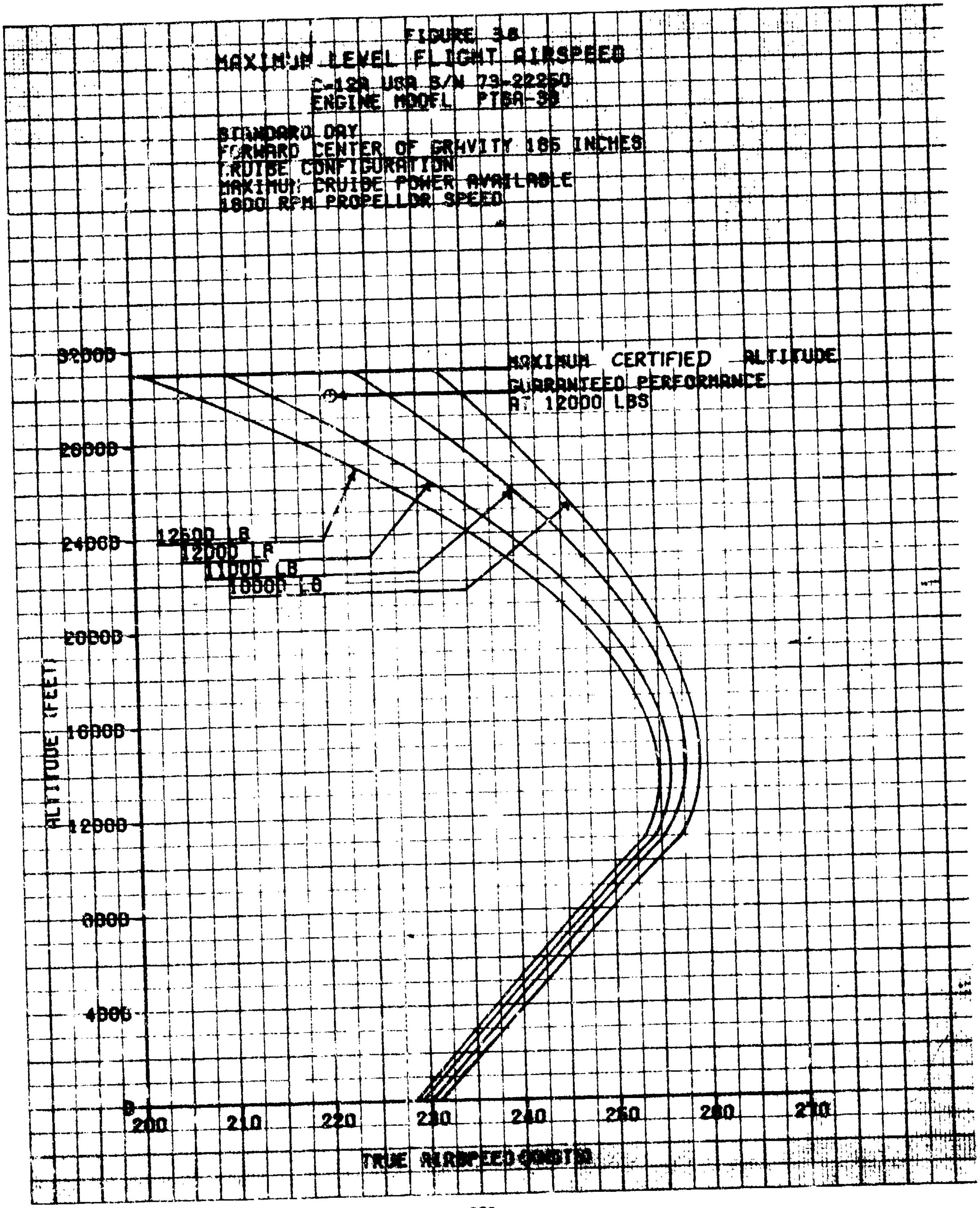


FIGURE 38
MAXIMUM LEVEL FLIGHT AIRSPEED
C-12A USA S/N 73-22250
ENGINE MODEL PT6A-38
STANDARD DAY
FORWARD CENTER OF GRAVITY 185 INCHES
CRUISE CONFIGURATION
MAXIMUM CRUISE POWER AVAILABLE
1800 RPM PROPELLOR SPEED
MAXIMUM CERTIFIED ALTITUDE
GUARANTEED PERFORMANCE
AT 12000 LBS
12500 LB
12000 LB
11000 LB
10000 LB
ALTITUDE (FEET)
32000
28000
24000
20000
16000
12000
8000
4000
0
200
210
220
230
240
250
260
270
TRUE AIRSPEED (KNOTS)

FIGURE 39
LEVEL FLIGHT RANGE SUMMARY
C-12A USA S/N 73-22250
ENGINE MODEL PT6A-38

CRUISE CONFIGURATION
STANDARD DAY CONDITIONS
FORWARD CENTER OF GRAVITY
PROPELLER SPEED = 1800 RPM
LONG RANGE CRUISE TRUE AIRSPEED DEFINED AT 0.99 OF MAXIMUM NAUTICAL AIRMILES PER POUND OF FUEL OR MAXIMUM LEVEL FLIGHT AIRSPEED

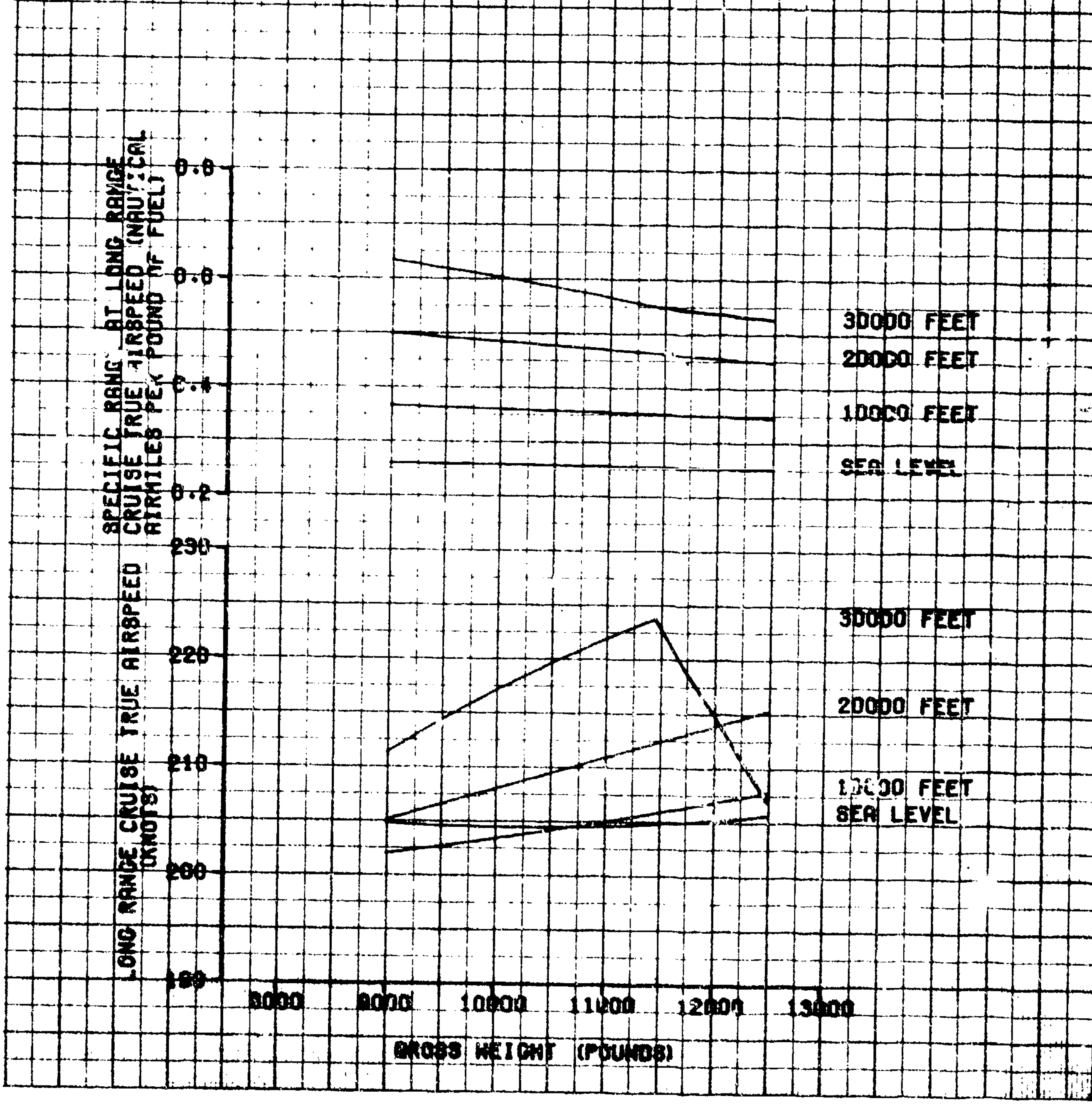


FIGURE 40
MAXIMUM ENDURANCE SUMMARY
C-12A USA S/N 73-22250
ENGINE MODEL PT6A-38

STANDARD DAY CONDITIONS
FORWARD CENTER OF GRAVITY
1800 RPM PROPELLER SPEED
SPECIFICATION FUEL FLOW BASED ON UACL COMPUTER DATA

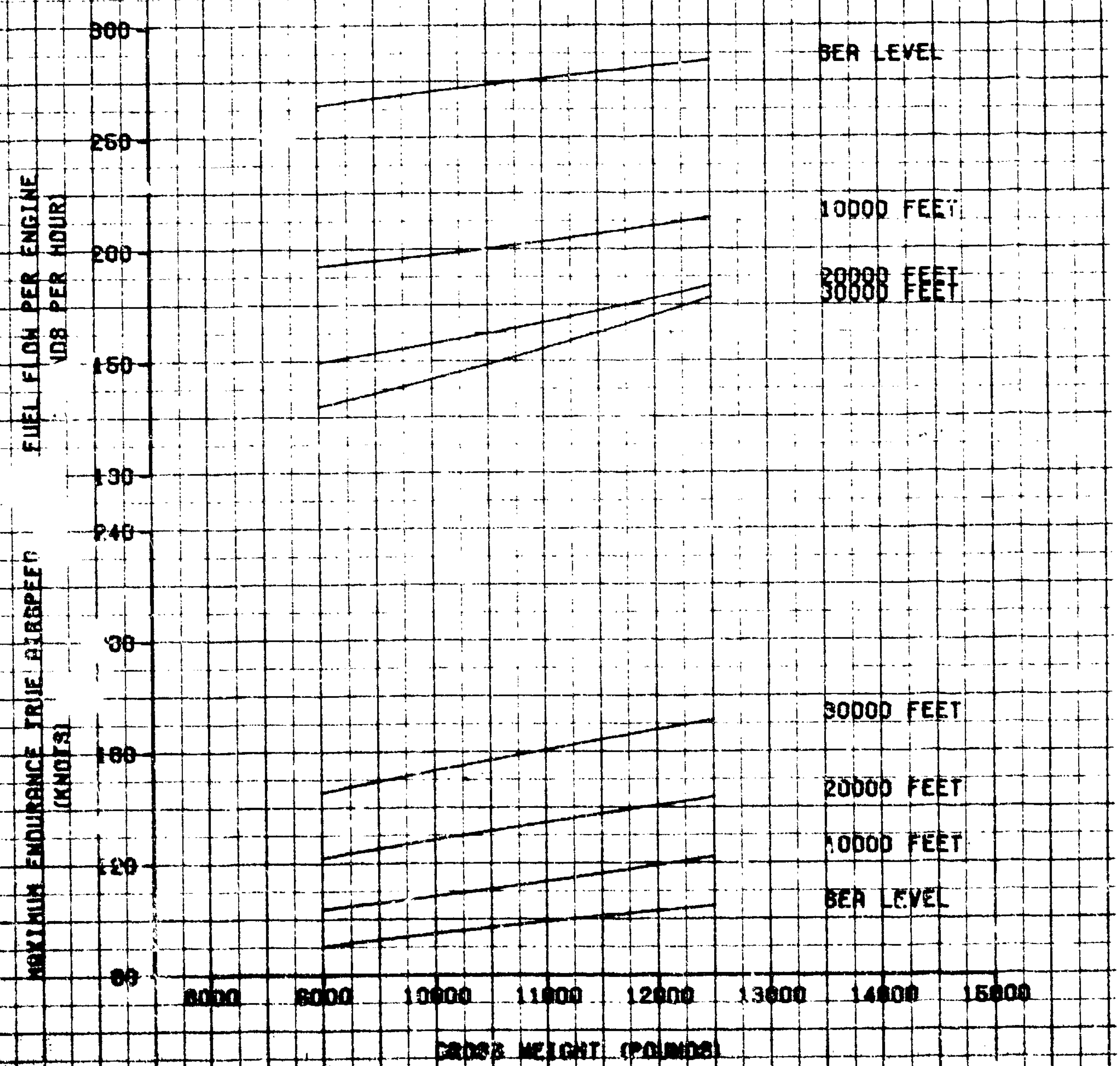
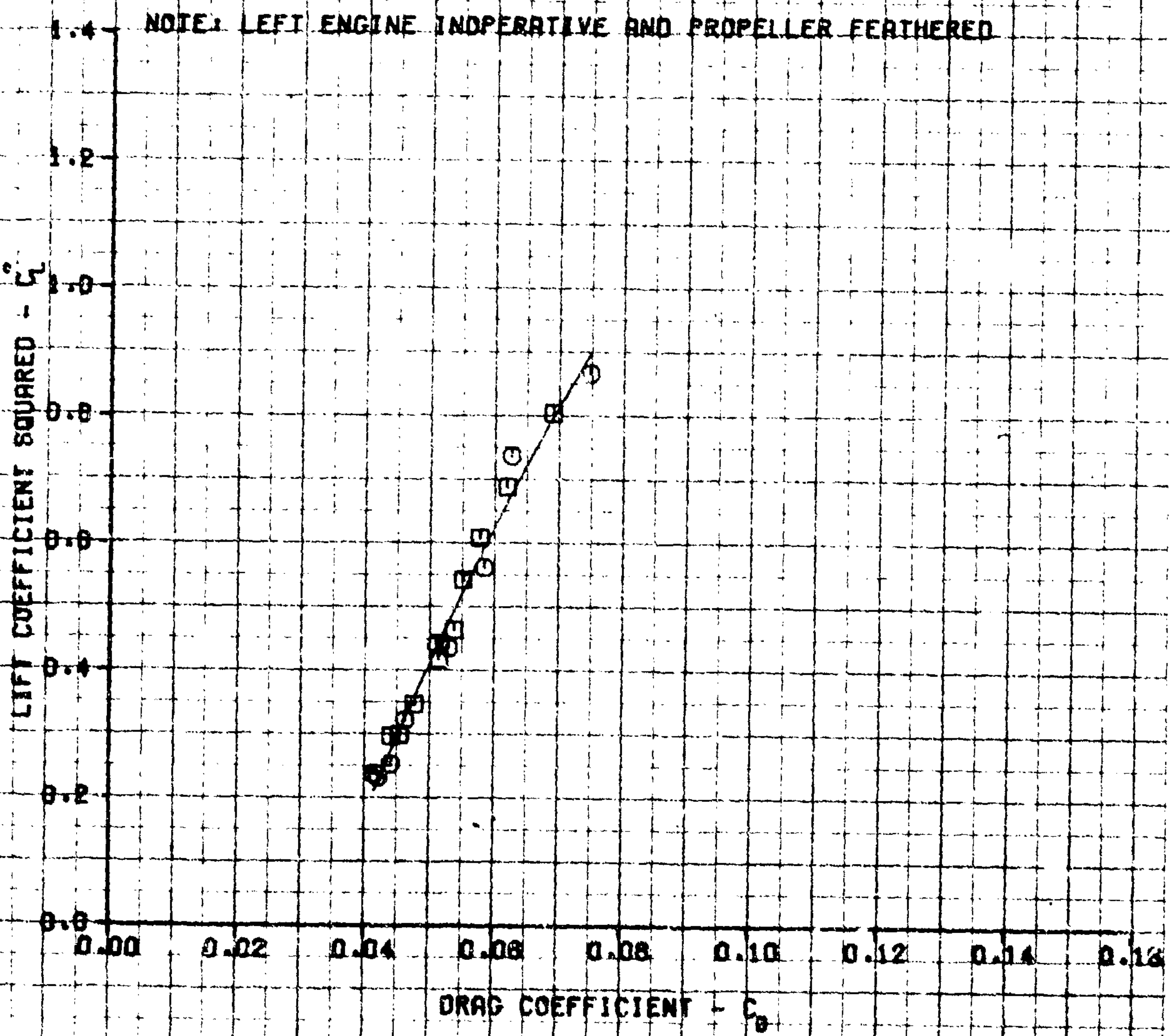


FIGURE 41

SINGLE ENGINE LEVEL FLIGHT DRAG POLAR

C-12A USA S/N 73-22250

SYM	AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	PROPELLER SPEED (RPM)	CONFIGURATION	FLIGHT CONDITION
○	12660.	186.2(FWD)	5690.	12.0	1800.	CRUISE	LEVEL FLIGHT
□	12580.	186.2(FWD)	11520.	9.0	1800.	CRUISE	LEVEL FLIGHT



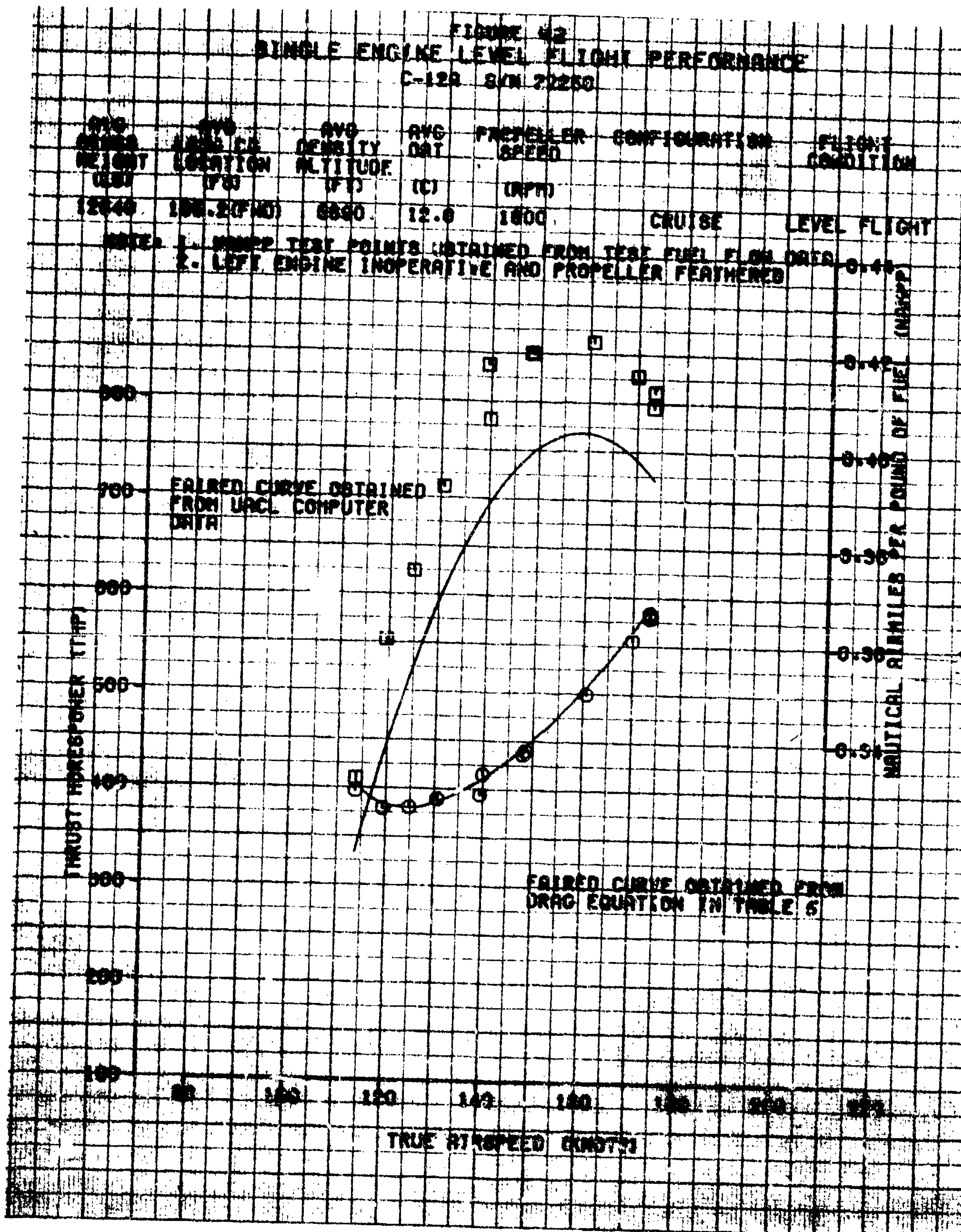

SINGLE ENGINE LEVEL FLIGHT PERFORMANCE
C-12A
AVG DENSITY ALTITUDE (FT)
AVG OAT (C)
PROPELLER SPEED (RPM)
CONFIGURATION
FLIGHT CONDITION
12.0
1600
CRUISE
LEVEL FLIGHT
2. LEFT ENGINE INOPERATIVE AND PROPELLER FEATHERED
FAIRED CURVE OBTAINED FROM UACL COMPUTER DATA
FAIRED CURVE OBTAINED FROM DRAG EQUATION IN TABLE 5
THRUST HORSEPOWER (THP)
NAUTICAL AIRMILES PER POUND OF FUEL (NAMPP)
700
600
500
400
300
200
100
120
140
160
TRUE AIRSPEED (KNOTS)

FIGURE 43
SINGLE ENGINE LEVEL FLIGHT PERFORMANCE
C-12A S/N 22250

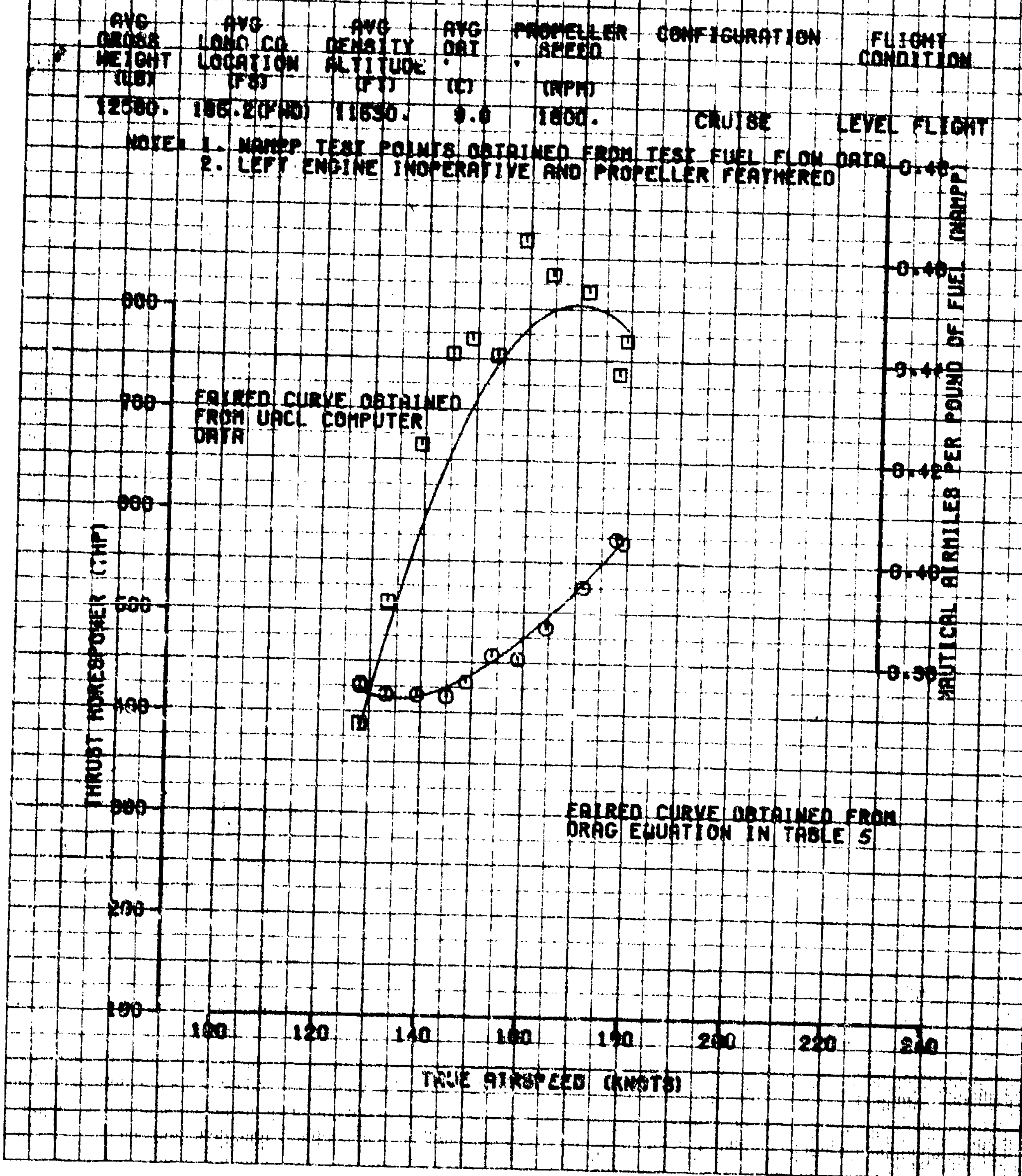
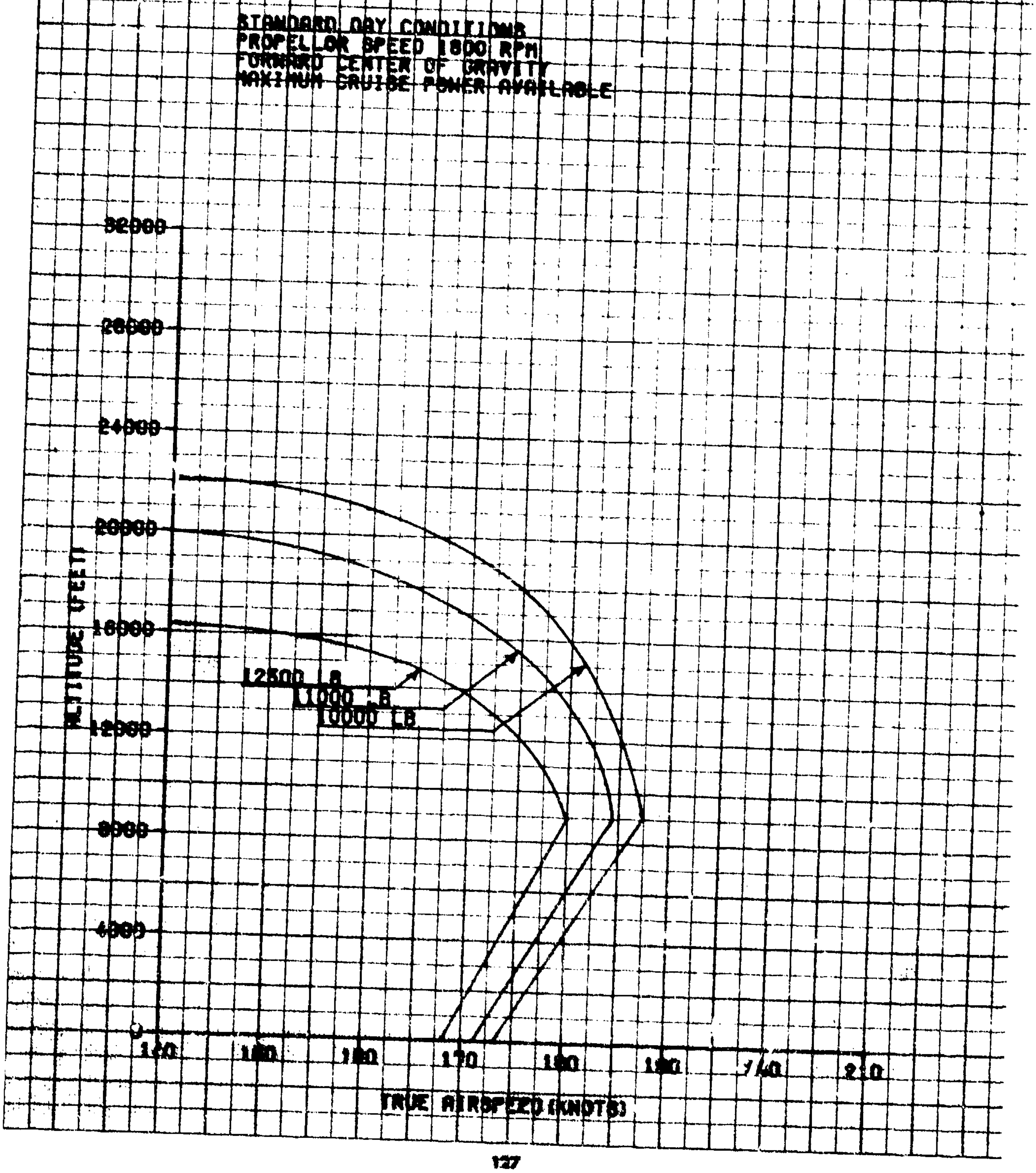


FIGURE 44
SINGLE ENGINE MAXIMUM LEVEL FLIGHT AIRSPEED
C-12A USA S/N 73-22250



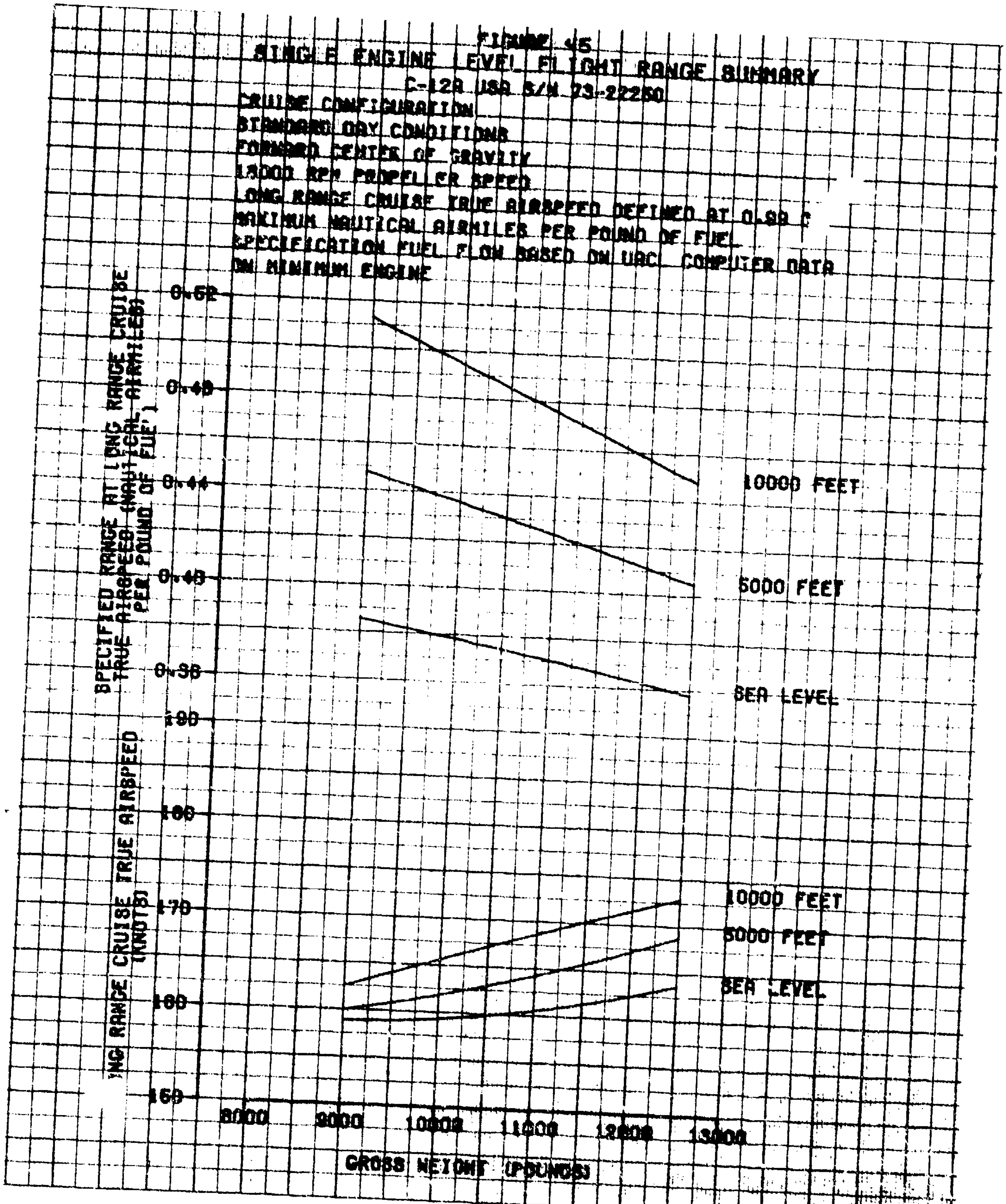


FIGURE 45
SINGLE ENGINE LEVEL FLIGHT RANGE SUMMARY
C-12A USA S/N 73-22250
CRUISE CONFIGURATION
STANDARD DAY CONDITIONS
FORWARD CENTER OF GRAVITY
1700 RPM PROPELLER SPEED
LONG RANGE CRUISE TRUE AIRSPEED DEFINED AT 0.99 OF
MAXIMUM NAUTICAL AIRMILES PER POUND OF FUEL
SPECIFICATION FUEL FLOW BASED ON UACL COMPUTER DATA
ON MINIMUM ENGINE
SPECIFIED RANGE AT LONG RANGE CRUISE TRUE AIRSPEED (NAUTICAL AIRMILES PER POUND OF FUEL)
0.52
0.48
0.44
0.40
0.36
10000 FEET
5000 FEET
SEA LEVEL
LONG RANGE CRUISE TRUE AIRSPEED (KNOTS)
190
180
170
160
150
10000 FEET
5000 FEET
SEA LEVEL
8000
9000
10000
11000
12000
13000
GROSS WEIGHT (POUNDS)

FIGURE 46

SINGLE ENGINE MAXIMUM ENDURANCE SUMMARY

C-12A USA S/N 73-22260
ENGINE MODEL PT6A-38

STANDARD DAY CONDITIONS
FORWARD CENTER OF GRAVITY
1800 RPM PROPELLER SPEED
SPECIFICATION FUEL FLOW BASED ON UACL COMPUTER DATA

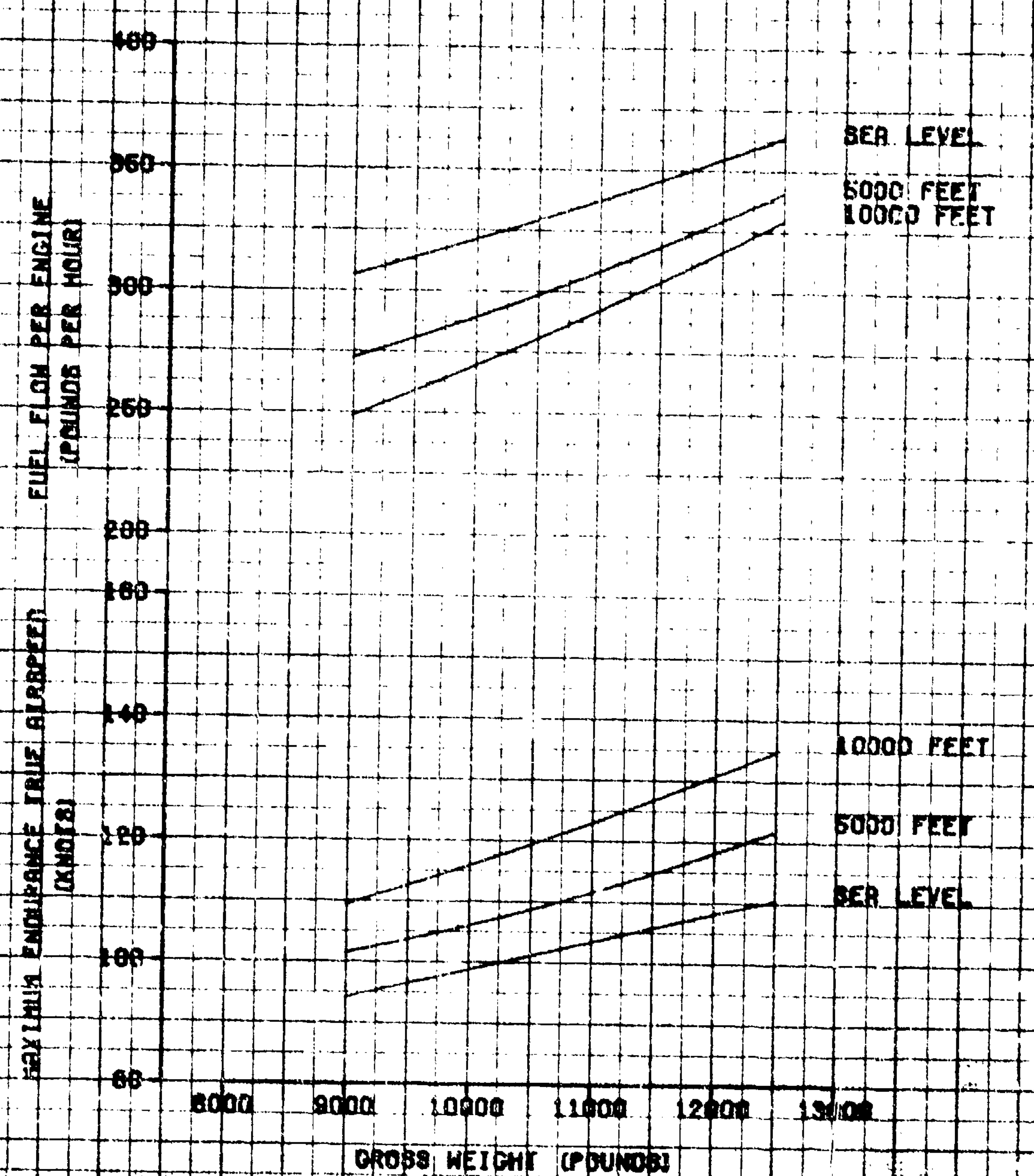


FIGURE 47
UNACCELERATED STALL
C-12A USA S/N 73-22250

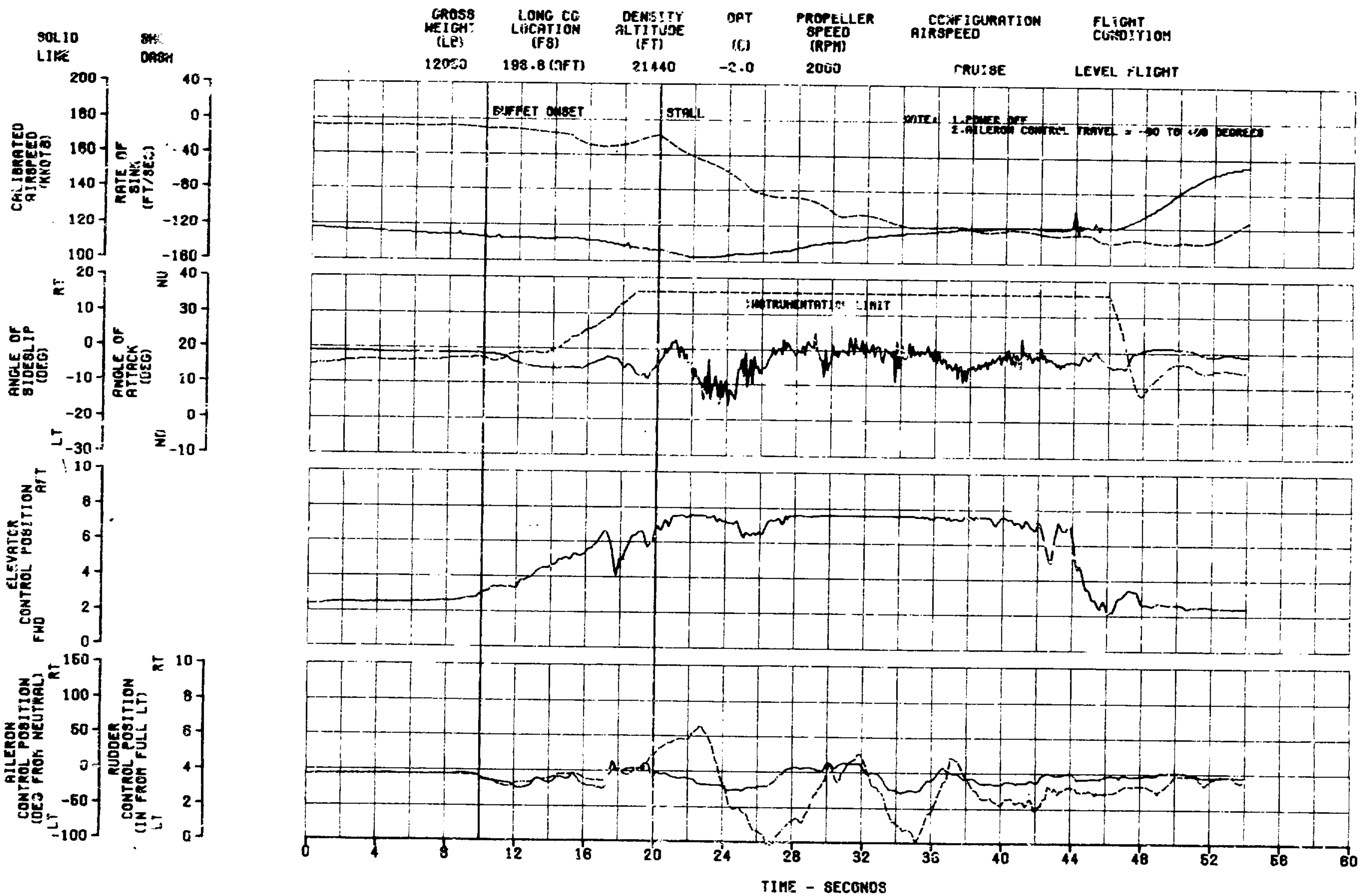


FIGURE 48
UNACCELERATED STALL
C-12A USA S/N 73-22250

GROSS WEIGHT (LB)	LONG CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (C)	PROPELLER SPEED (RPM)	CONFIGURATION	FLIGHT CONDITION
12720	196.4 (AFT)	9800	-2.5	2000	CRUISE	LEVEL FLIGHT

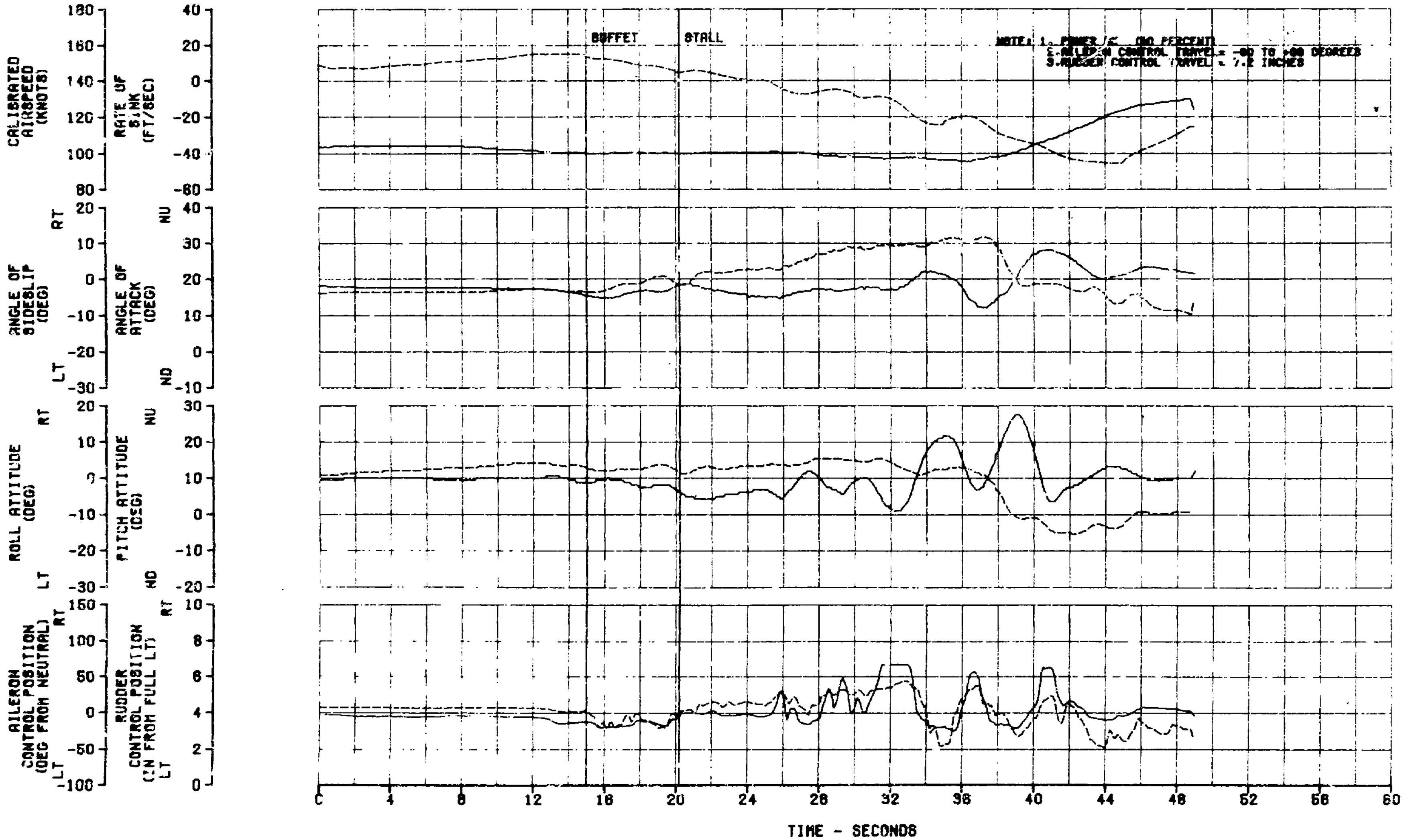
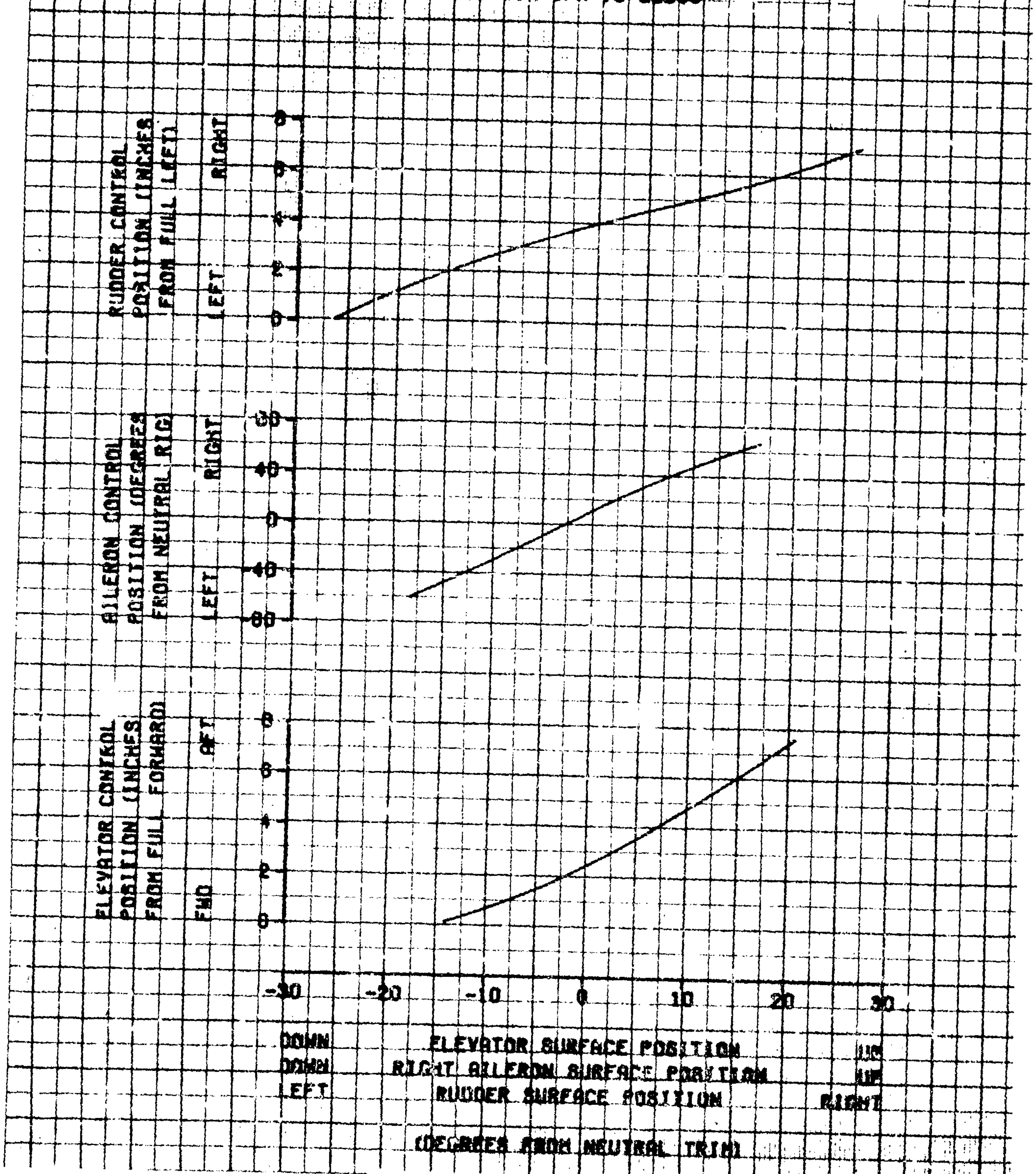


FIGURE 49
COCKPIT CONTROL/CONTROL SURFACE RELATIONSHIP
C-12A USA S/N 73-22250



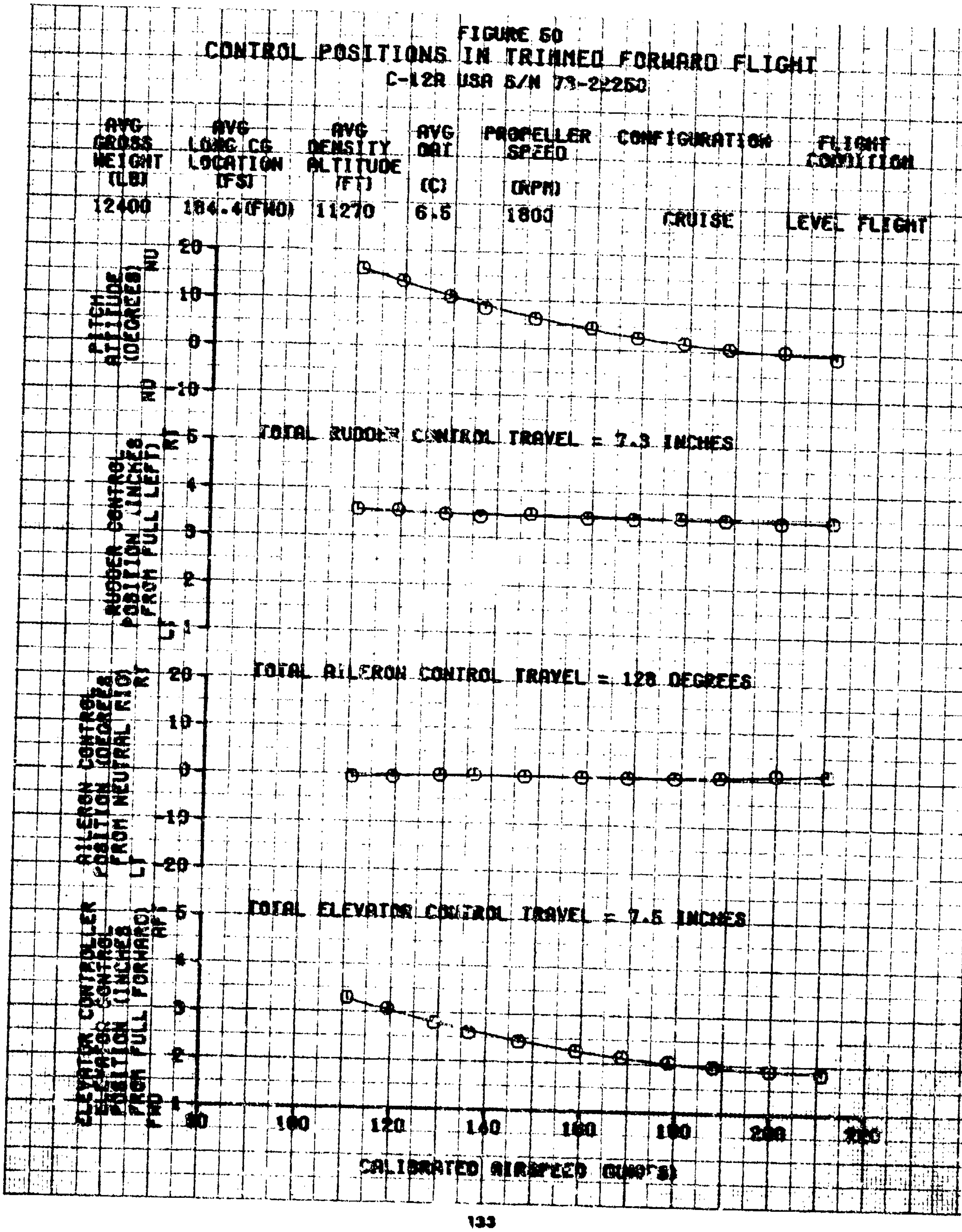


FIGURE 50
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
C-12A USA S/N 73-22250
AVG GROSS WEIGHT (LB) 12400
AVG LONG CG LOCATION (FS) 184.4(FWD)
AVG DENSITY ALTITUDE (FT) 11270
AVG OAT (C) 6.5
PROPELLER SPEED (RPM) 1800
CONFIGURATION CRUISE
FLIGHT CONDITION LEVEL FLIGHT
PITCH ATTITUDE (DEGREES)
NU
ND
TOTAL RUDDER CONTROL TRAVEL = 7.3 INCHES
RUDDER CONTROL POSITION (INCHES FROM FULL LEFT)
RT
LT
TOTAL AILERON CONTROL TRAVEL = 128 DEGREES
AILERON CONTROL POSITION (DEGREES FROM NEUTRAL RIG)
TOTAL ELEVATOR CONTROL TRAVEL = 7.5 INCHES
ELEVATOR CONTROL POSITION (INCHES FROM FULL FORWARD)
AFT
FWD
CALIBRATED AIRSPEED (KNOTS)

FIGURE 51

CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT

C-12A USA S/N 73-22250

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	PROPELLER SPEED (RPM)	CONFIGURATION	FLIGHT CONDITION
11880	186.2 (MID)	11120	3.5	1900	POWER APPROACH	DESCENT

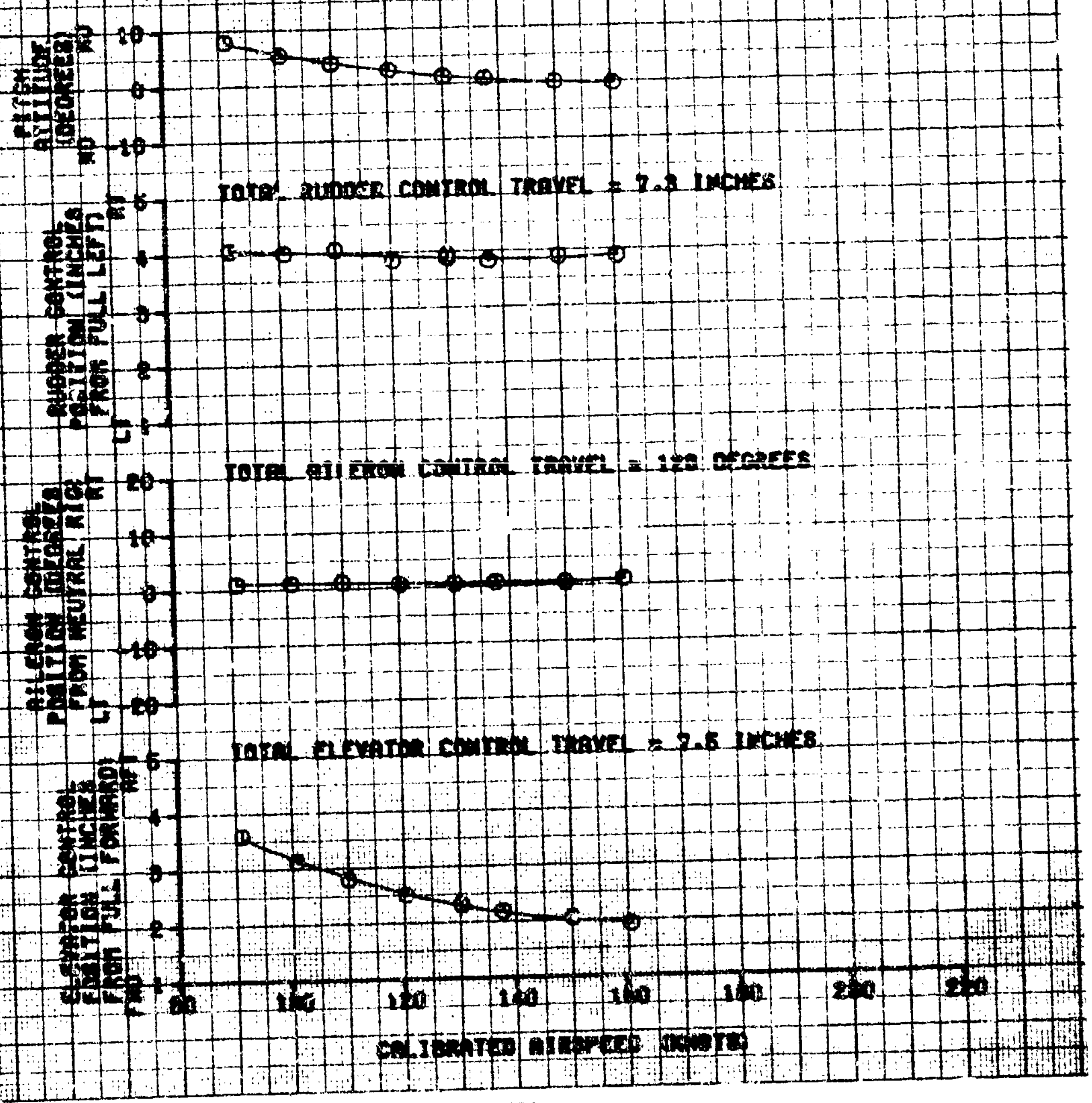
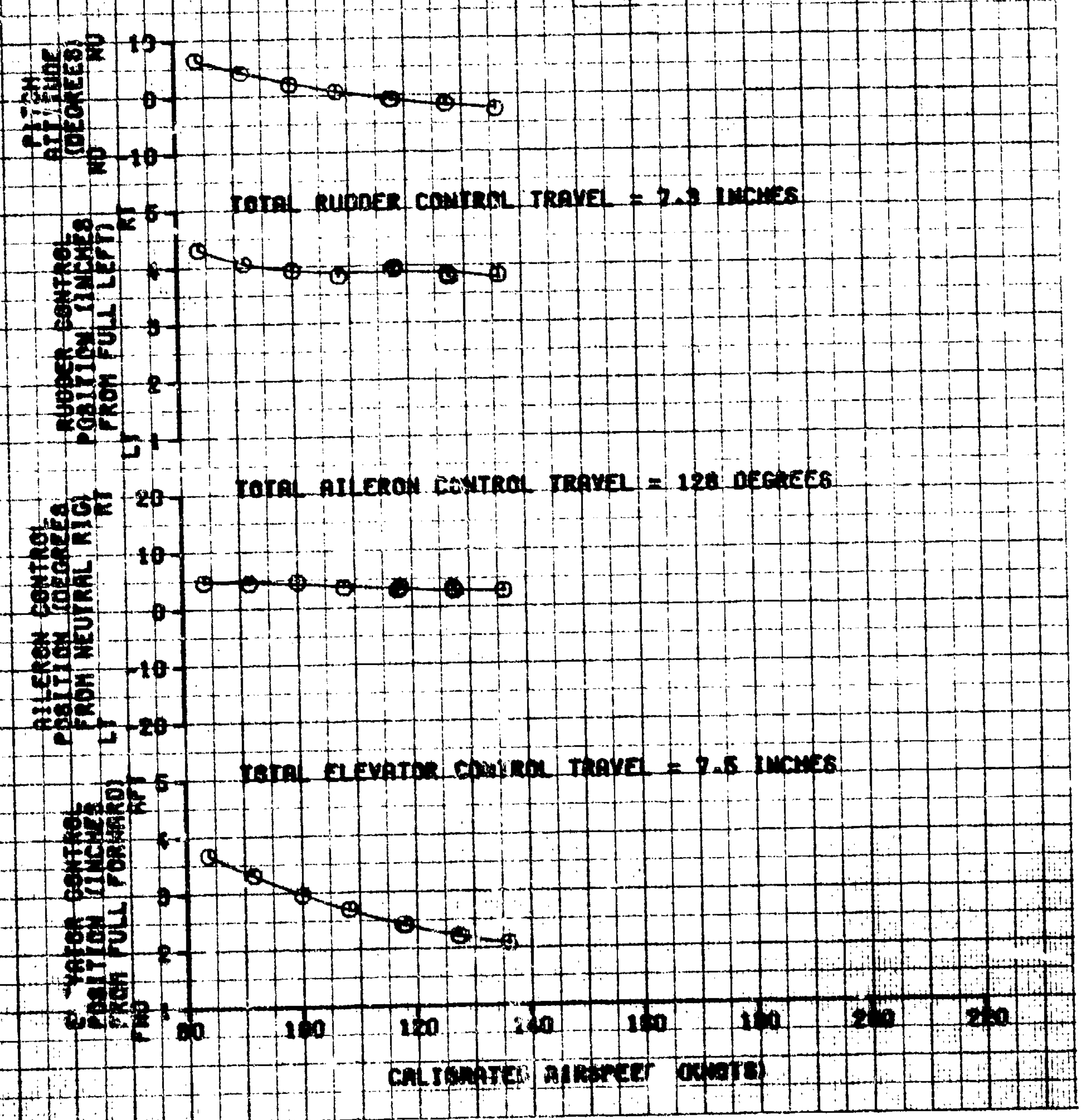


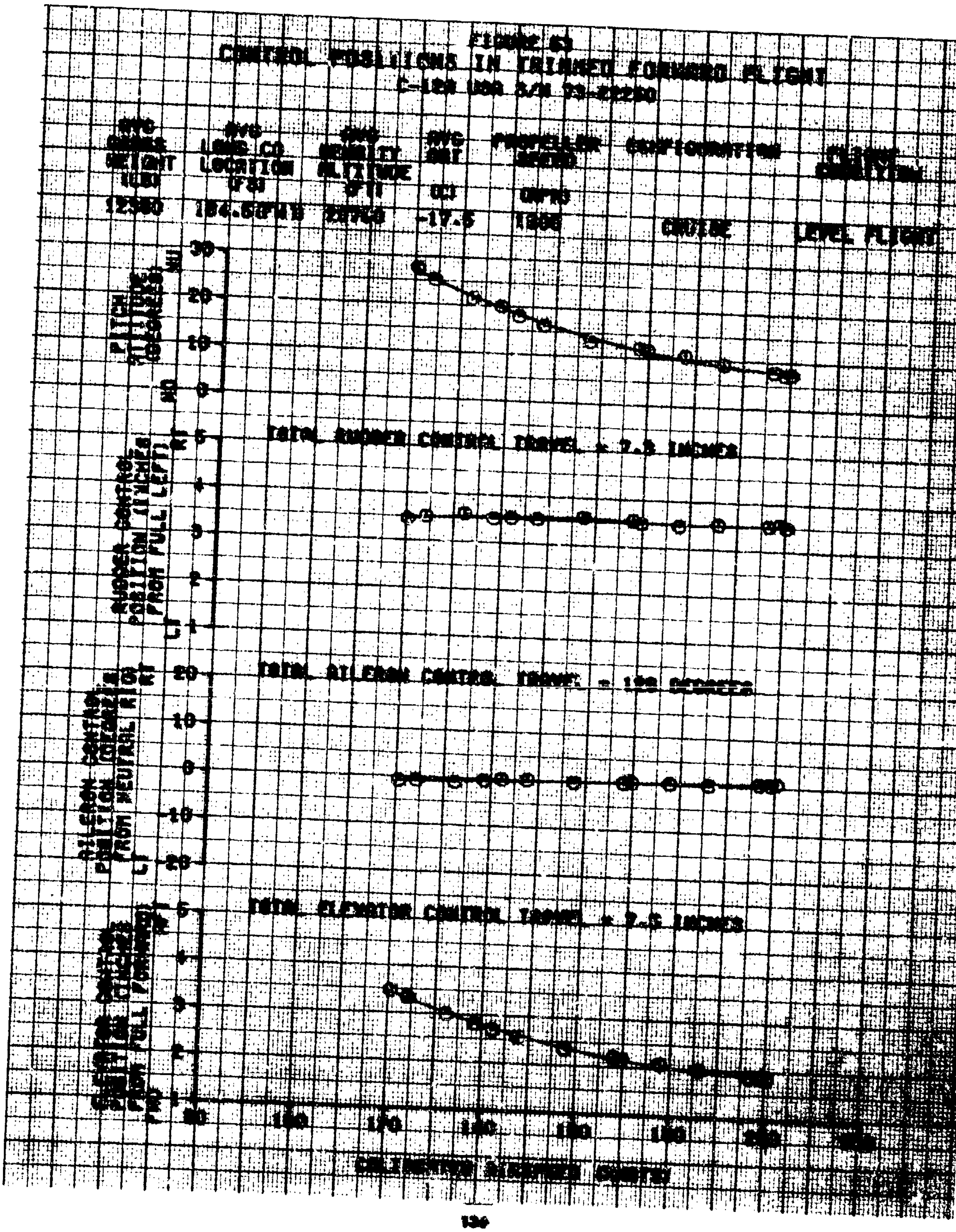
FIGURE 52

CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT

C-12A USA S/N 73-22250

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	PROPELLER SPEED (RPM)	CONFIGURATION	FLIGHT CONDITION
11580	185.2(FWD)	11060	3.5	1880	WAVE OFF	CLIMB




CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
CRUISE
LEVEL FLIGHT
PITCH ATTITUDE (DEGREES)
TOTAL RUDDER CONTROL TRAVEL = 7.5 INCHES
RUDDER CONTROL POSITION (INCHES FROM FULL LEFT)
TOTAL ELEVATOR CONTROL TRAVEL = 7.5 INCHES

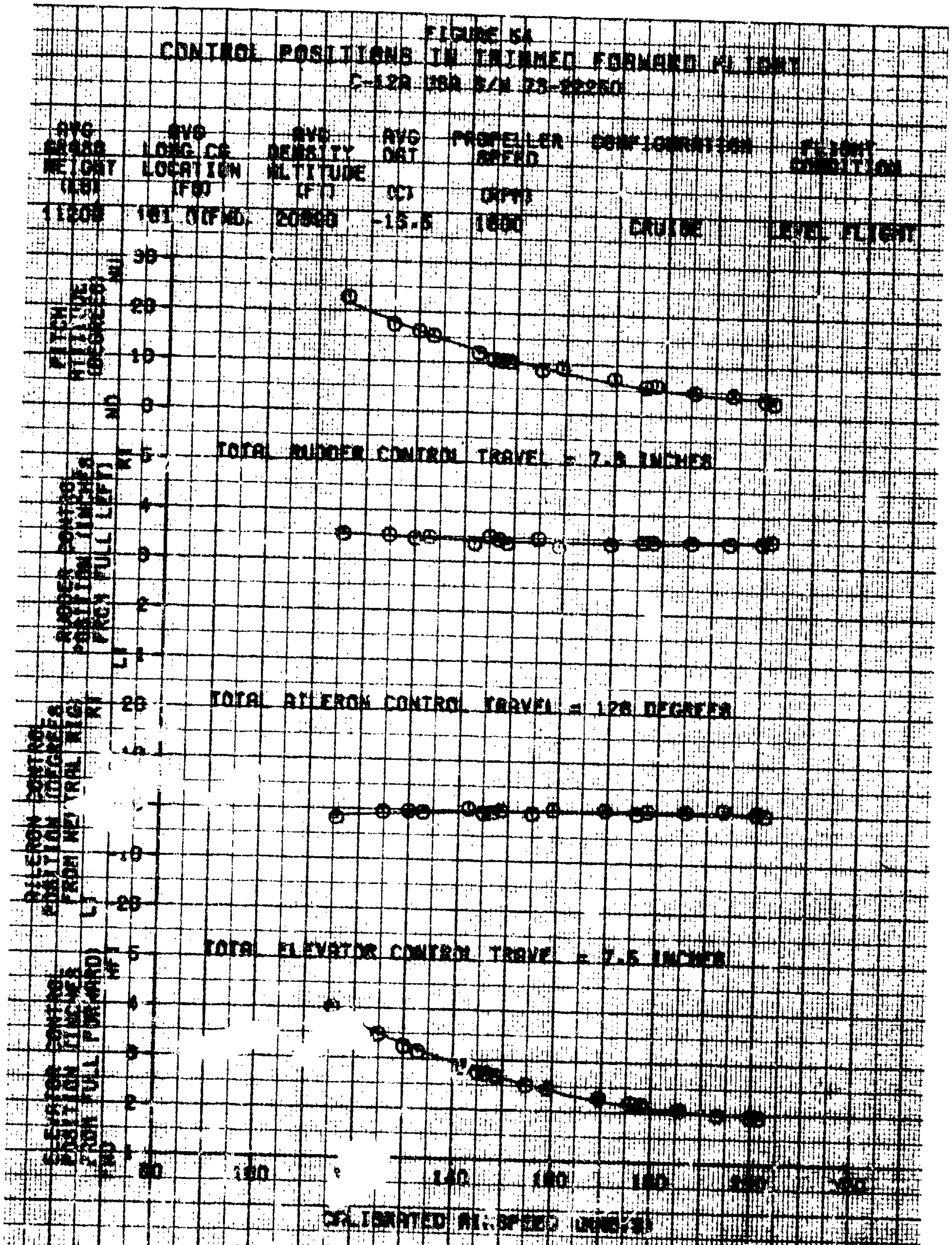


FIGURE 5A
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
C-12A USA S/N 73-22250
AVG GROSS WEIGHT (LB)
AVG LONG CG LOCATION (FS)
AVG DENSITY ALTITUDE (FT)
AVG OAT (C)
PROPELLER SPEED (RPM)
CONFIGURATION
FLIGHT CONDITION
11200
161.0(FWD)
20000
-15.5
1800
CRUISE
LEVEL FLIGHT
PITCH ATTITUDE (DEGREES)
TOTAL RUDDER CONTROL TRAVEL = 7.8 INCHES
RUDDER CONTROL POSITION (INCHES FROM FULL LEFT)
TOTAL AILERON CONTROL TRAVEL = 120 DEGREES
AILERON CONTROL POSITION (DEGREES FROM NEUTRAL RIGHT)
TOTAL ELEVATOR CONTROL TRAVEL = 7.5 INCHES
ELEVATOR CONTROL POSITION (INCHES FROM FULL FORWARD)
CALIBRATED AIRSPEED (KNOTS)

FIGURE 56

CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT

C-12A USA S/N 73-22250

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	PROPELLER SPEED (RPM)	CONFIGURATION	FLIGHT CONDITION
12340	190.7(AFT)	21460	-12.7	1800	CRUISE	LEVEL FLIGHT

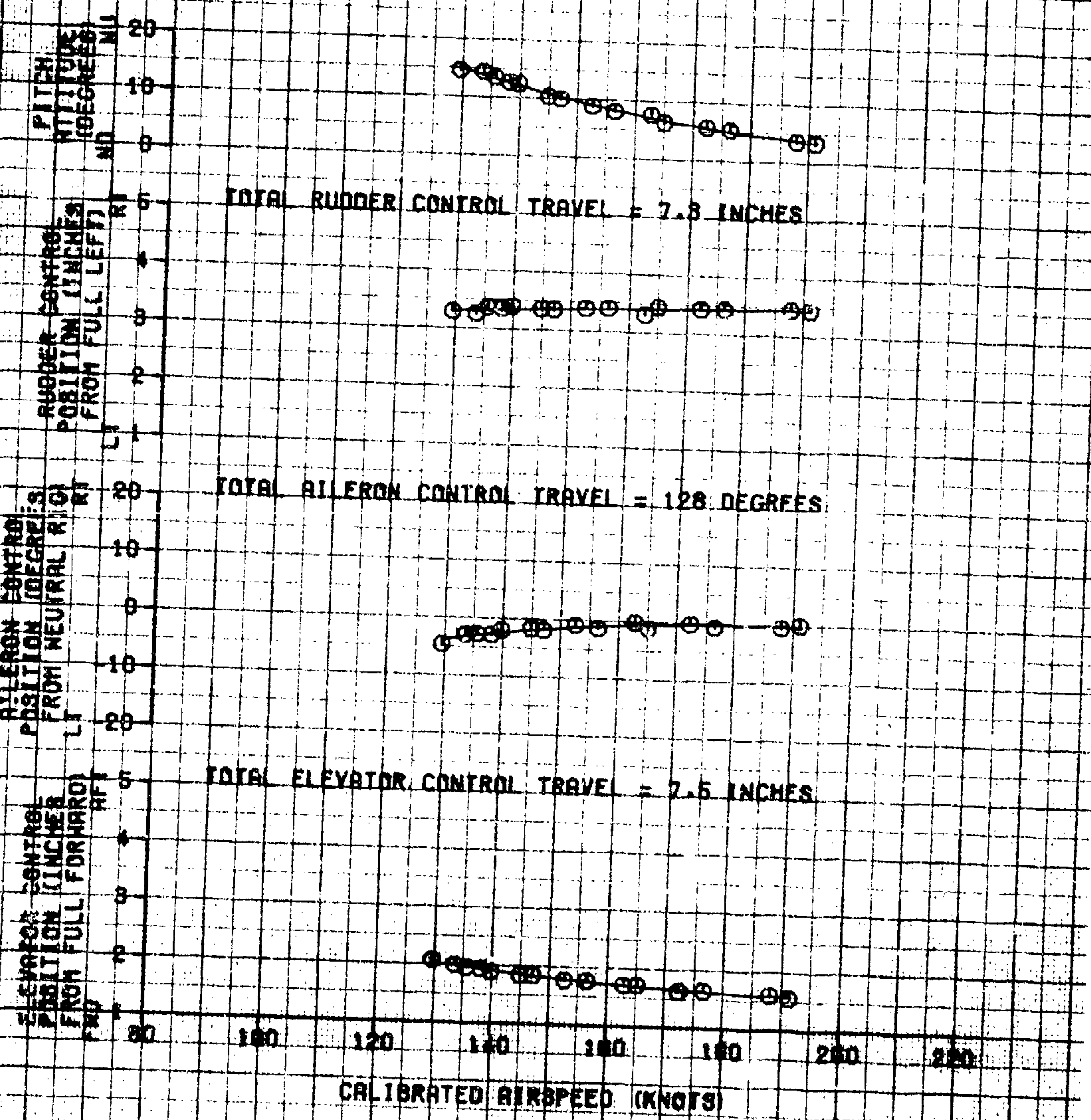
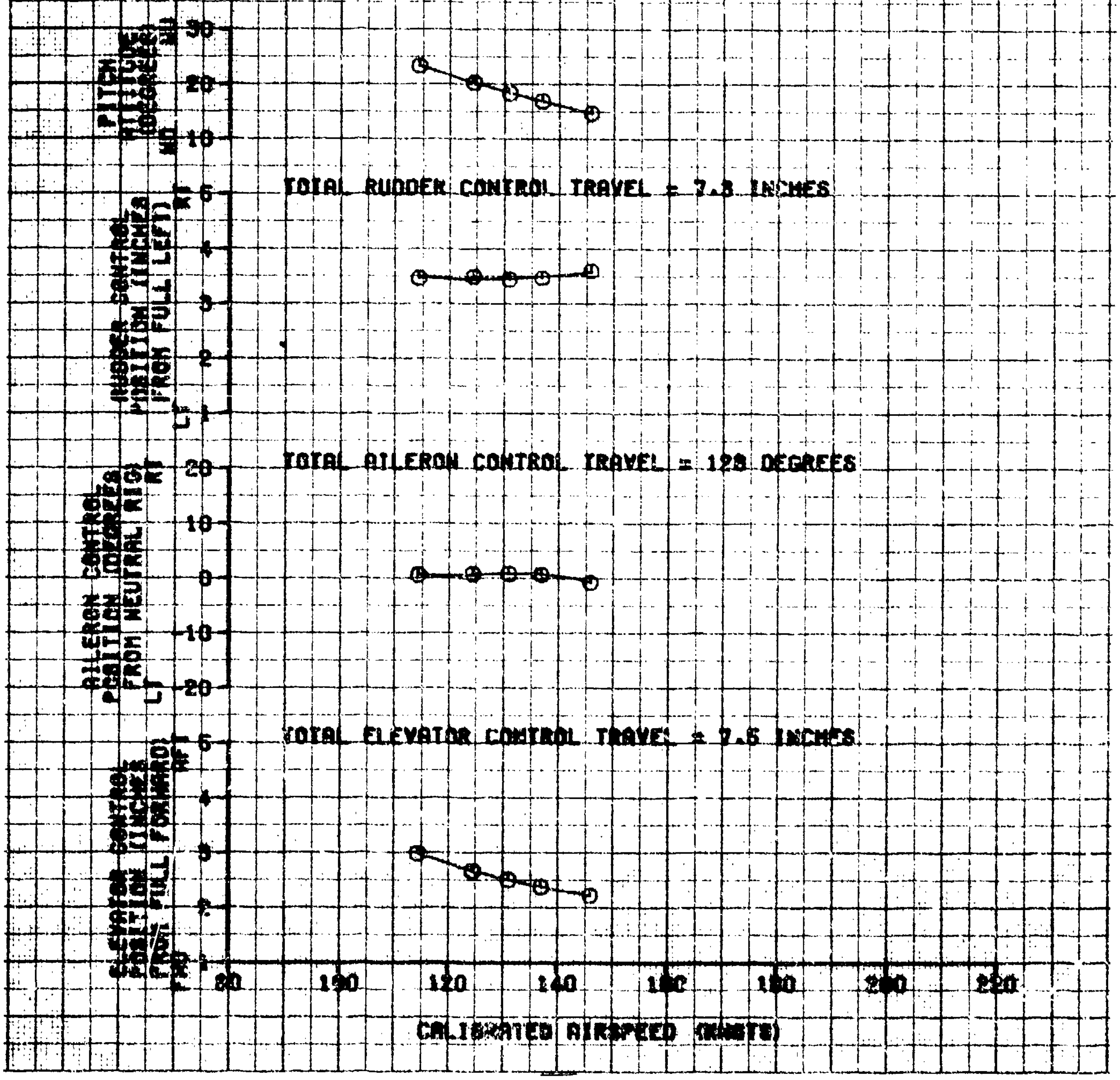


FIGURE 58

CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT

C-12C USA S/N 73-22250

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	PROPELLER SPEED (RPM)	CONFIGURATION	FLIGHT CONDITION
11300	186.0(FWD)	30800	-30.5	1800	CRUISE	LEVEL FLIGHT



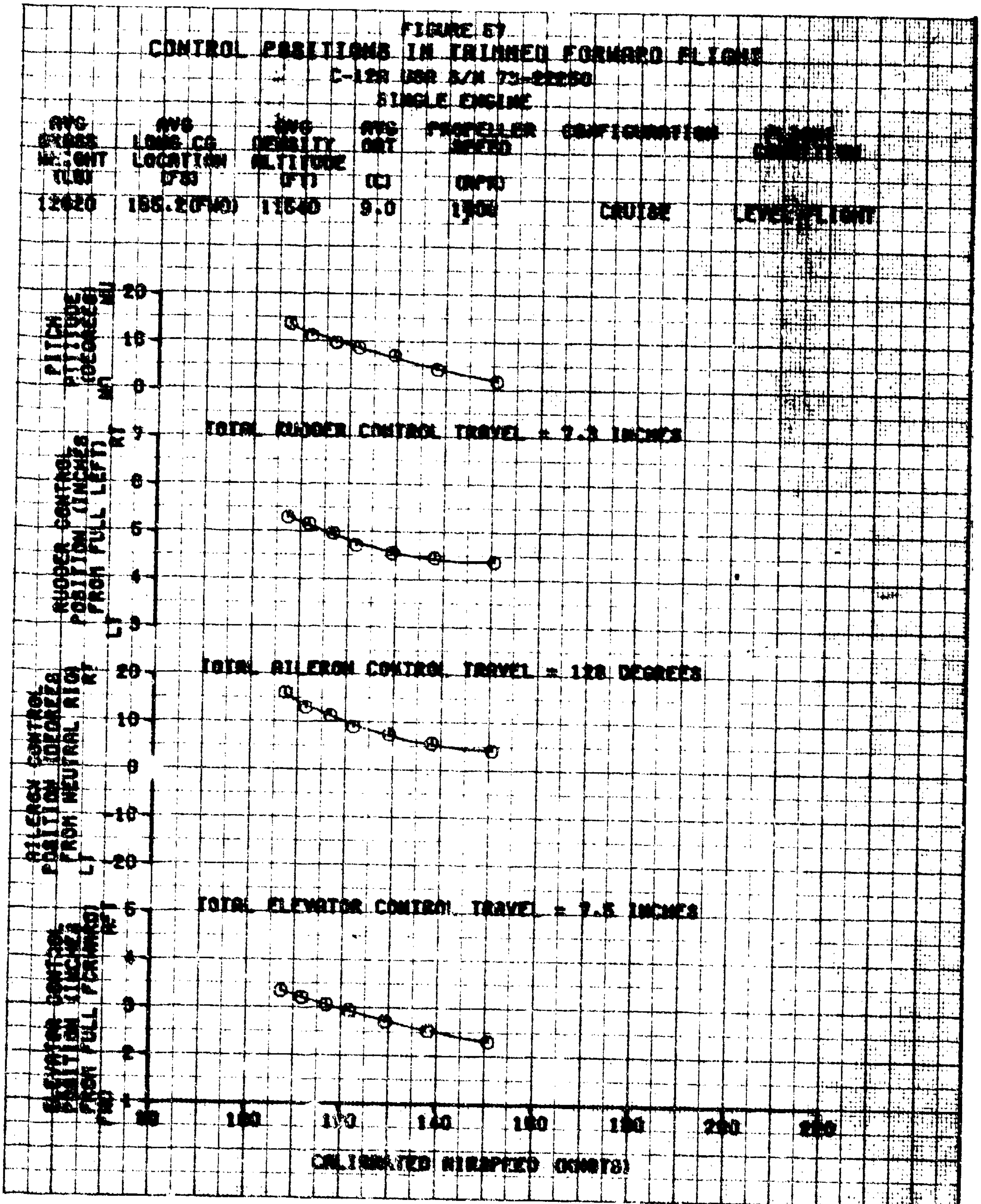


FIGURE 57
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
SINGLE ENGINE
AVG GROSS WEIGHT (LB)
AVG LONG CG LOCATION (FS)
AVG DENSITY ALTITUDE (FT)
AVG OAT (C)
PROPELLER SPEED (RPM)
CONFIGURATION
FLIGHT CONDITION
12620
185.2(FWD)
11540
9.0
CRUISE
LEVEL FLIGHT
PITCH ATTITUDE (DEGREES)
TOTAL RUDDER CONTROL TRAVEL = 7.3 INCHES
RUDDER CONTROL POSITION (INCHES FROM FULL LEFT)
TOTAL AILERON CONTROL TRAVEL = 128 DEGREES
AILERON CONTROL POSITION (DEGREES FROM NEUTRAL RIGHT)
TOTAL ELEVATOR CONTROL TRAVEL = 7.5 INCHES
ELEVATOR CONTROL POSITION (INCHES FROM FULL FORWARD)
CALIBRATED AIRSPEED (KNOTS)

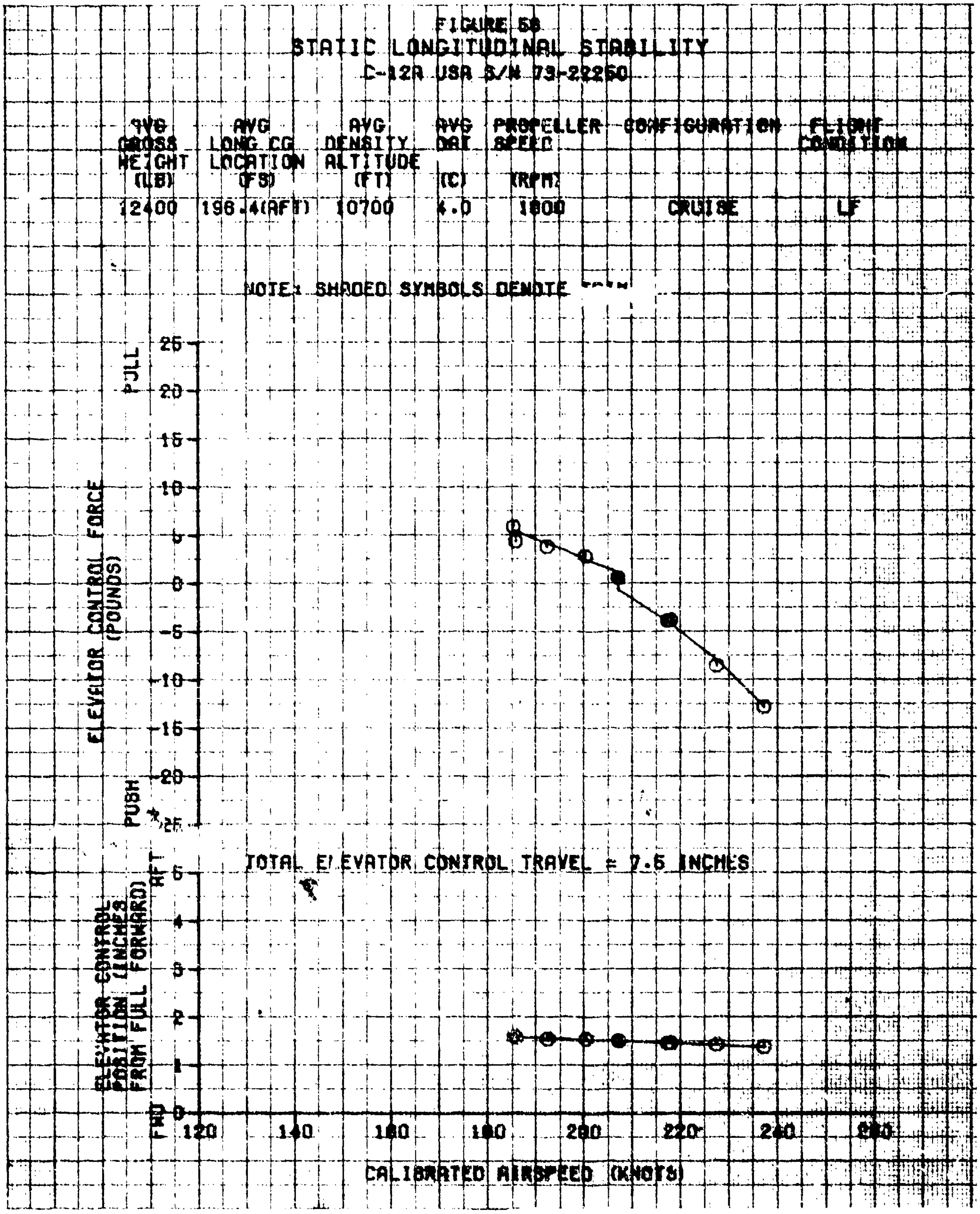


FIGURE 58
STATIC LONGITUDINAL STABILITY
C-12A USA S/N 73-22250
AVG GROSS WEIGHT (LB)
AVG LONG CG LOCATION (FS)
AVG DENSITY ALTITUDE (FT)
AVG OAT (C)
PROPELLER SPEED (RPM)
CONFIGURATION
FLIGHT CONDITION
12400
196.4(AFT)
10700
4.0
1800
CRUISE
LF
NOTE: SHADED SYMBOLS DENOTE TRIM
ELEVATOR CONTROL FORCE (POUNDS)
PULL
PUSH
25
20
15
10
5
0
-5
-10
-15
-20
-25
TOTAL ELEVATOR CONTROL TRAVEL = 7.5 INCHES
ELEVATOR CONTROL POSITION (INCHES FROM FULL FORWARD)
AFT
FWD
6
4
3
2
1
0
120
140
160
180
200
220
240
260
CALIBRATED AIRSPEED (KNOTS)

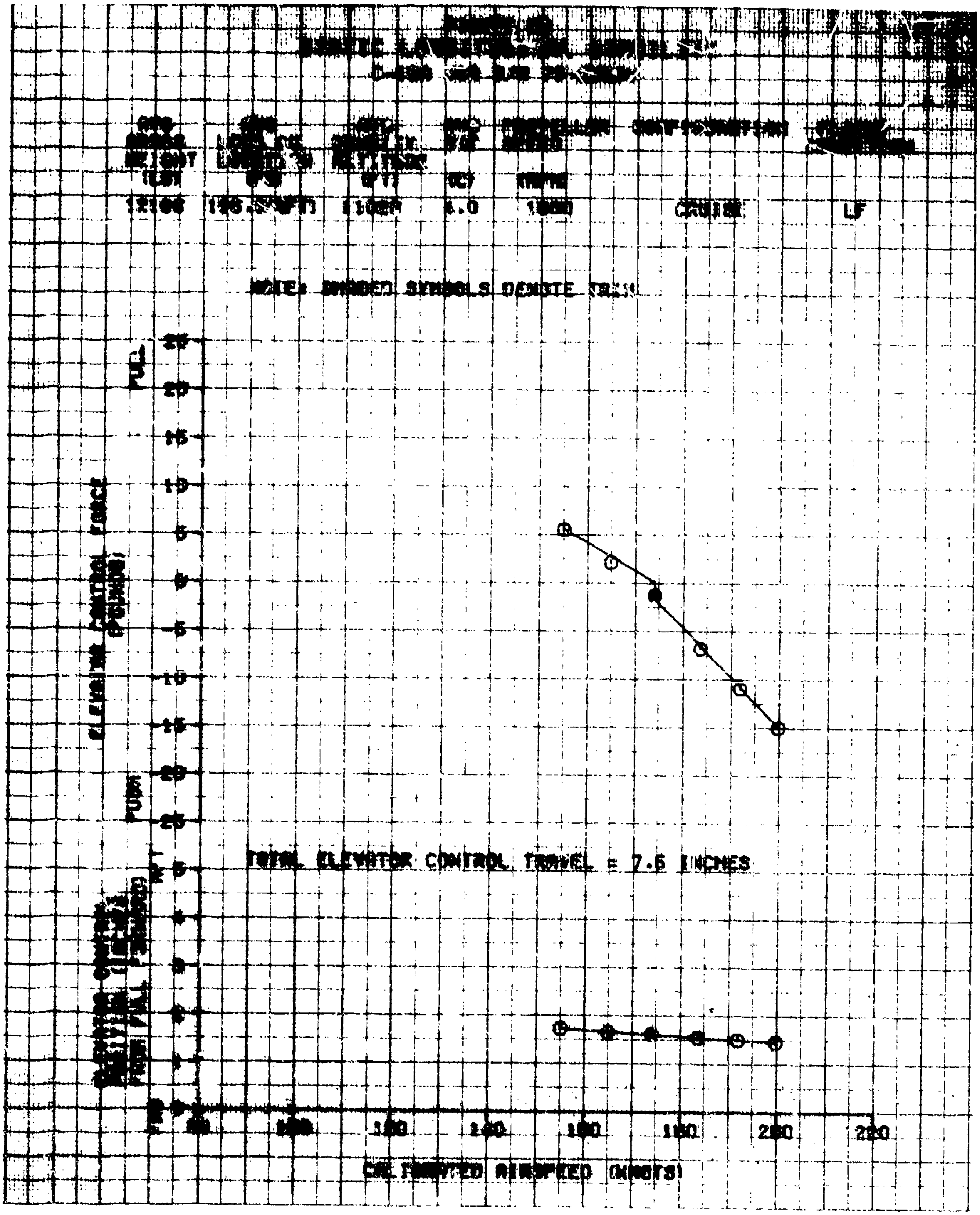

NOTE: SHADED SYMBOLS DENOTE TRIM
PULL
PUSH
ELEVATOR CONTROL FORCE (POUNDS)
TOTAL ELEVATOR CONTROL TRAVEL = 7.5 INCHES
CRUISE
LF
CALIBRATED AIRSPEED (KNOTS)

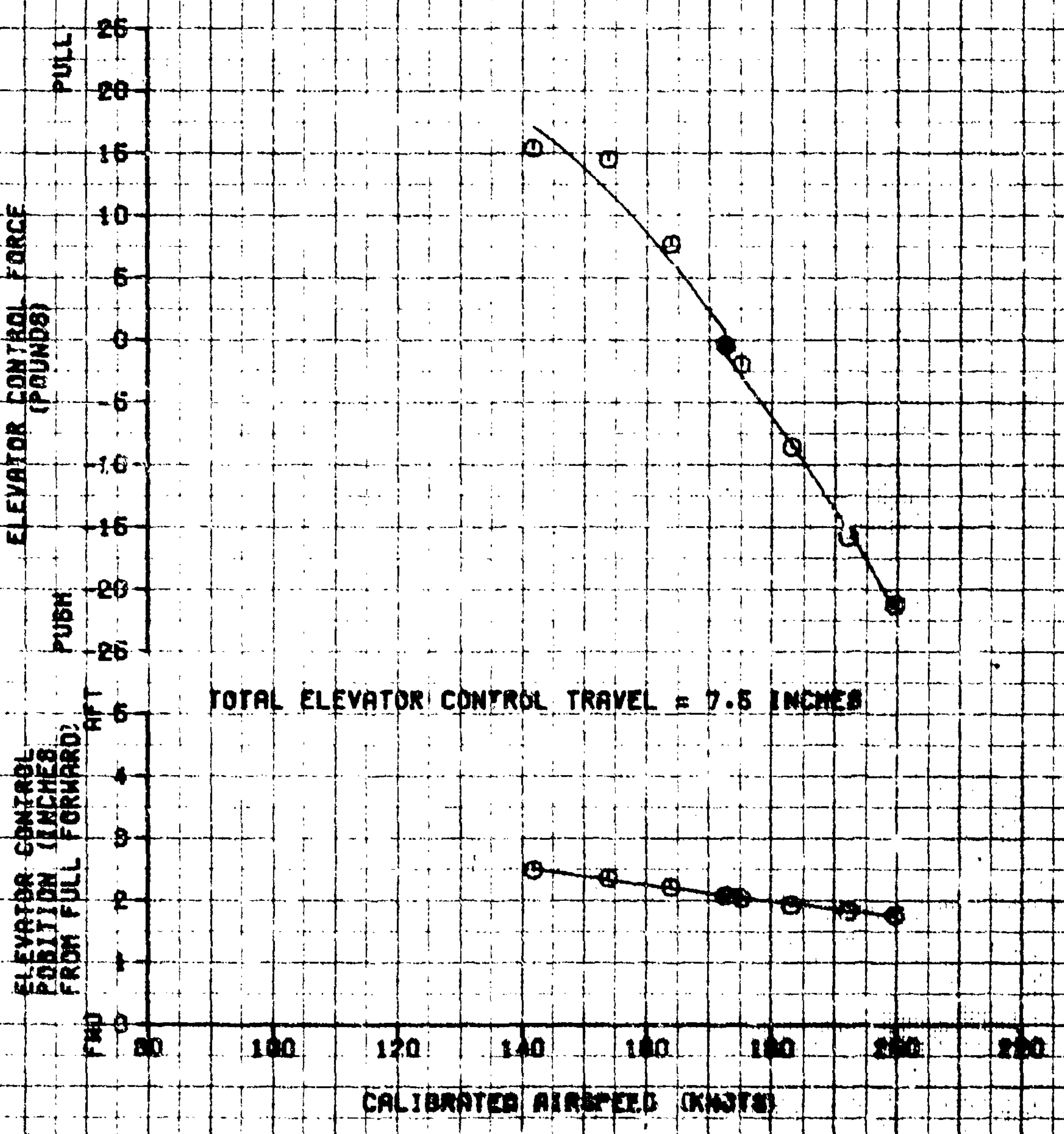
FIGURE 60

STATIC LONGITUDINAL STABILITY

C-12A USA S/N 73-22250

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	PROPELLER SPEED (RPM)	CONFIGURATION	FLIGHT CONDITION
12000	185.0(FWD)	11020	-1.5	1800	CRUISE	LF

NOTE: SHADED SYMBOLS DENOTE TRIM



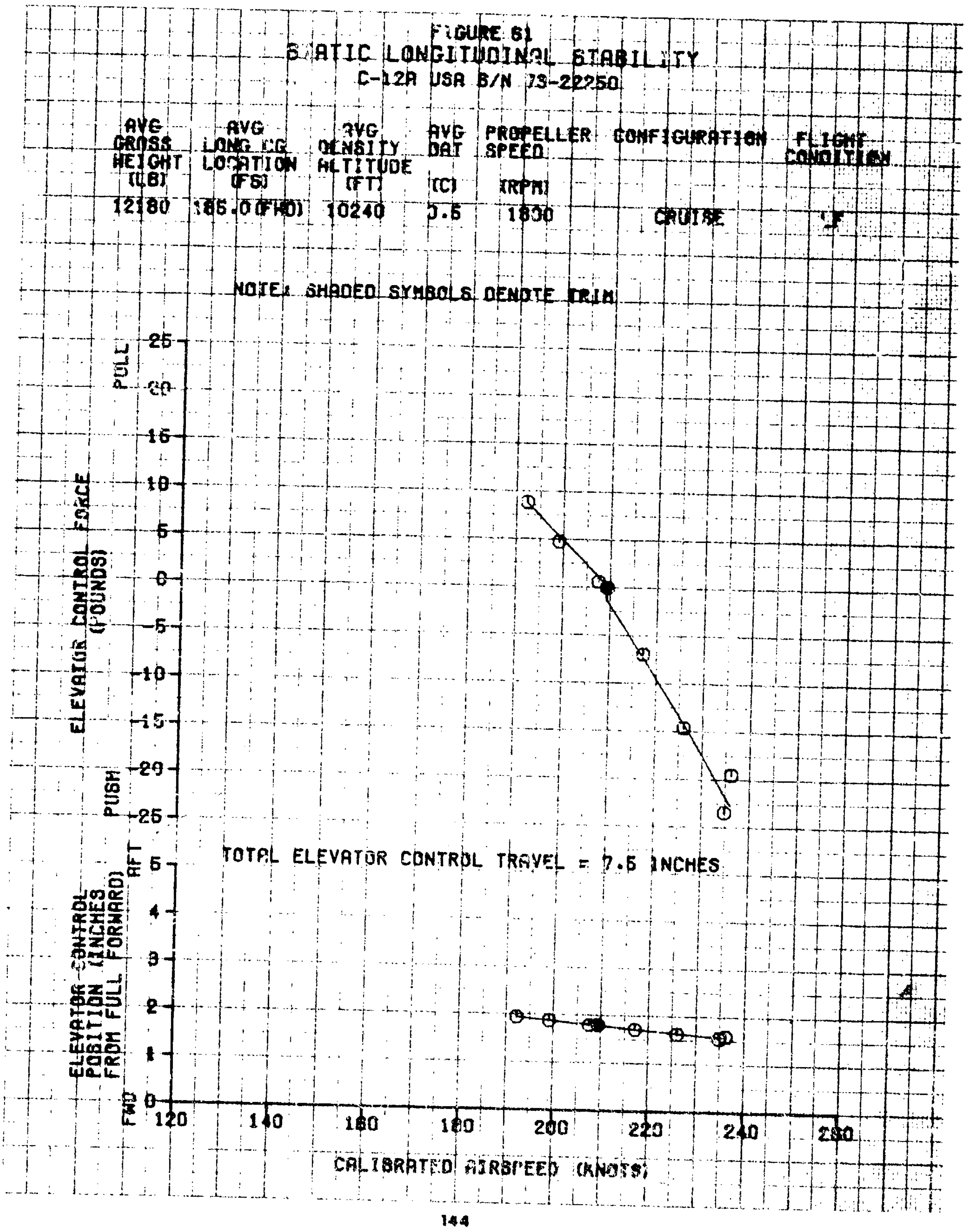


FIGURE 61
STATIC LONGITUDINAL STABILITY
C-12A USA S/N 73-22250
AVG GROSS WEIGHT (LB) 12180
AVG LONG CG LOCATION (FS) 186.0 (FWD)
AVG DENSITY ALTITUDE (FT) 10240
AVG OAT (C) 0.5
PROPELLER SPEED (RPM) 1800
CONFIGURATION CRUISE
FLIGHT CONDITION
NOTE: SHADED SYMBOLS DENOTE TRIM
ELEVATOR CONTROL FORCE (POUNDS)
PULL
PUSH
TOTAL ELEVATOR CONTROL TRAVEL = 7.5 INCHES
ELEVATOR CONTROL POSITION (INCHES FROM FULL FORWARD)
AFT
FWD
CALIBRATED AIRSPEED (KNOTS)

FIGURE 82
STATIC LONGITUDINAL STABILITY
C-12A USA S/N 73-22250

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	PROPELLER SPEED (RPM)	CONFIGURATION	FLIGHT CONDITION
12320	165.0 (FWD)	12530	-4.0	1900	CLIMB	CLIMB

NOTE: SHADED SYMBOLS DENOTE TRIM

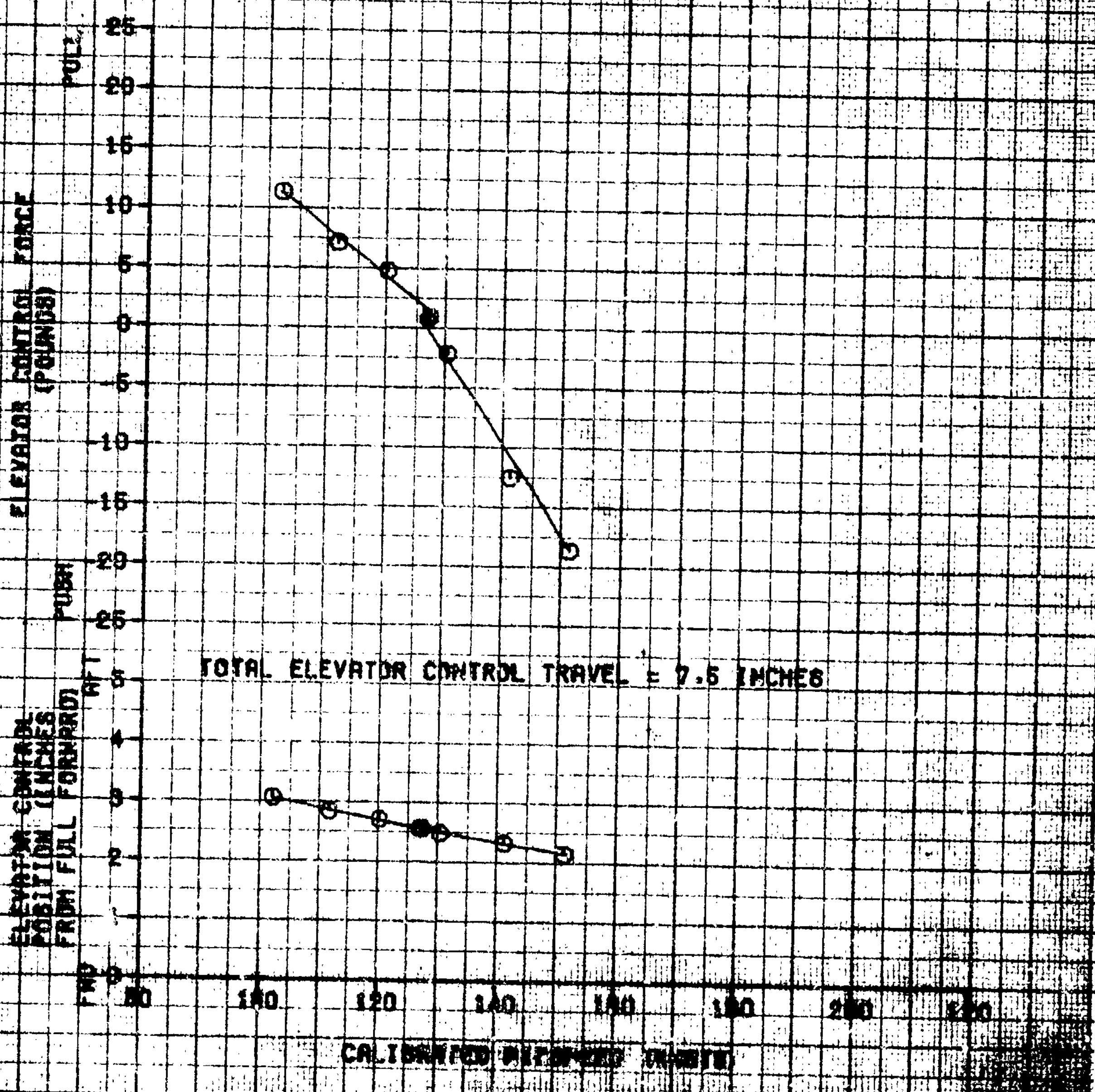


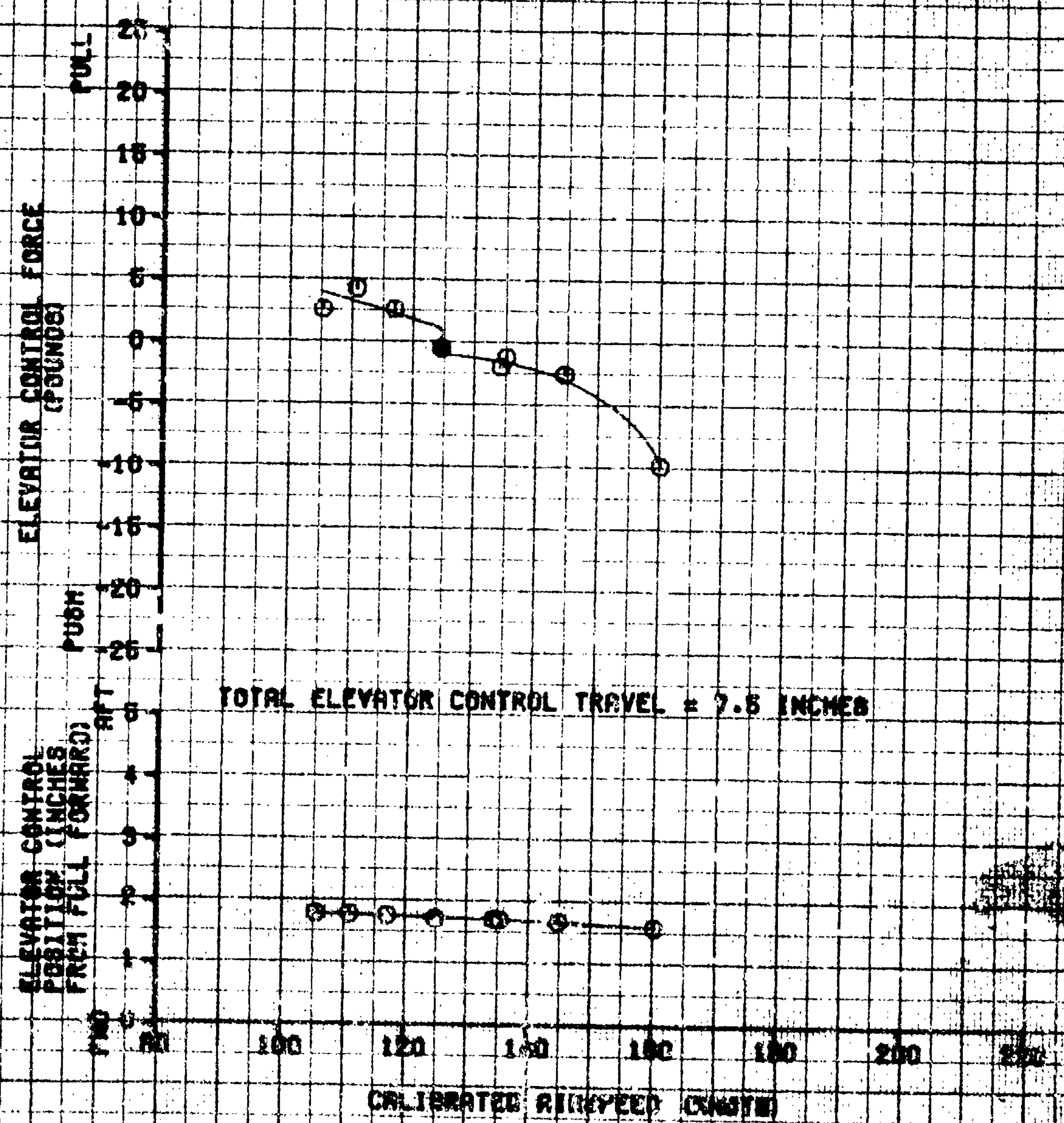
FIGURE 85

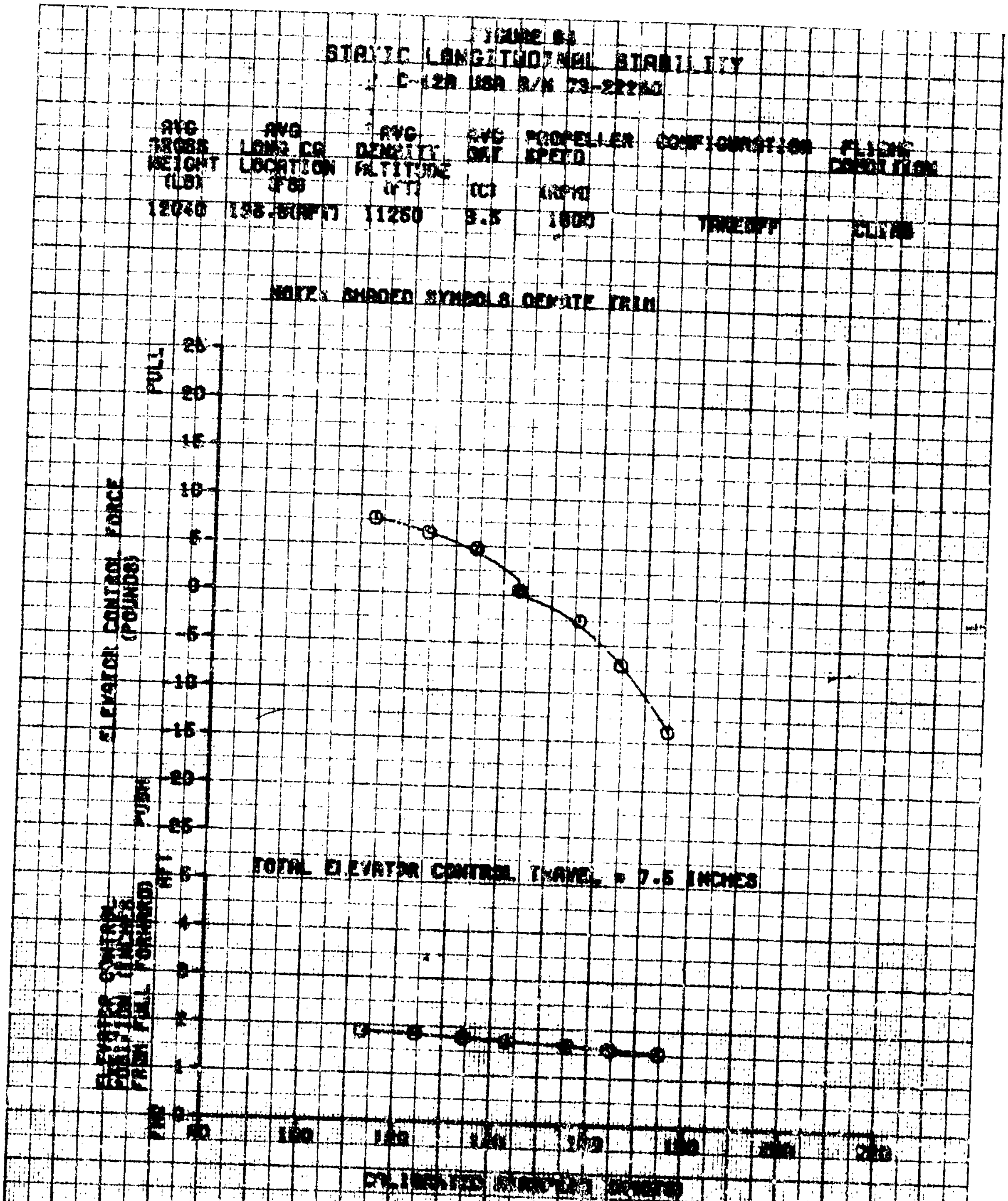
STATIC LONGITUDINAL STABILITY

C-12A USA S/N 73-22250

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	PROPELLER SPEED (RPM)	CONFIGURATION	FLIGHT CONDITION
12500	[illegible]1.5(AFT)	12950	-0.5	1800	CLIMB	CLIMB

NOTE: SHADED SYMBOLS DENOTE TRIM



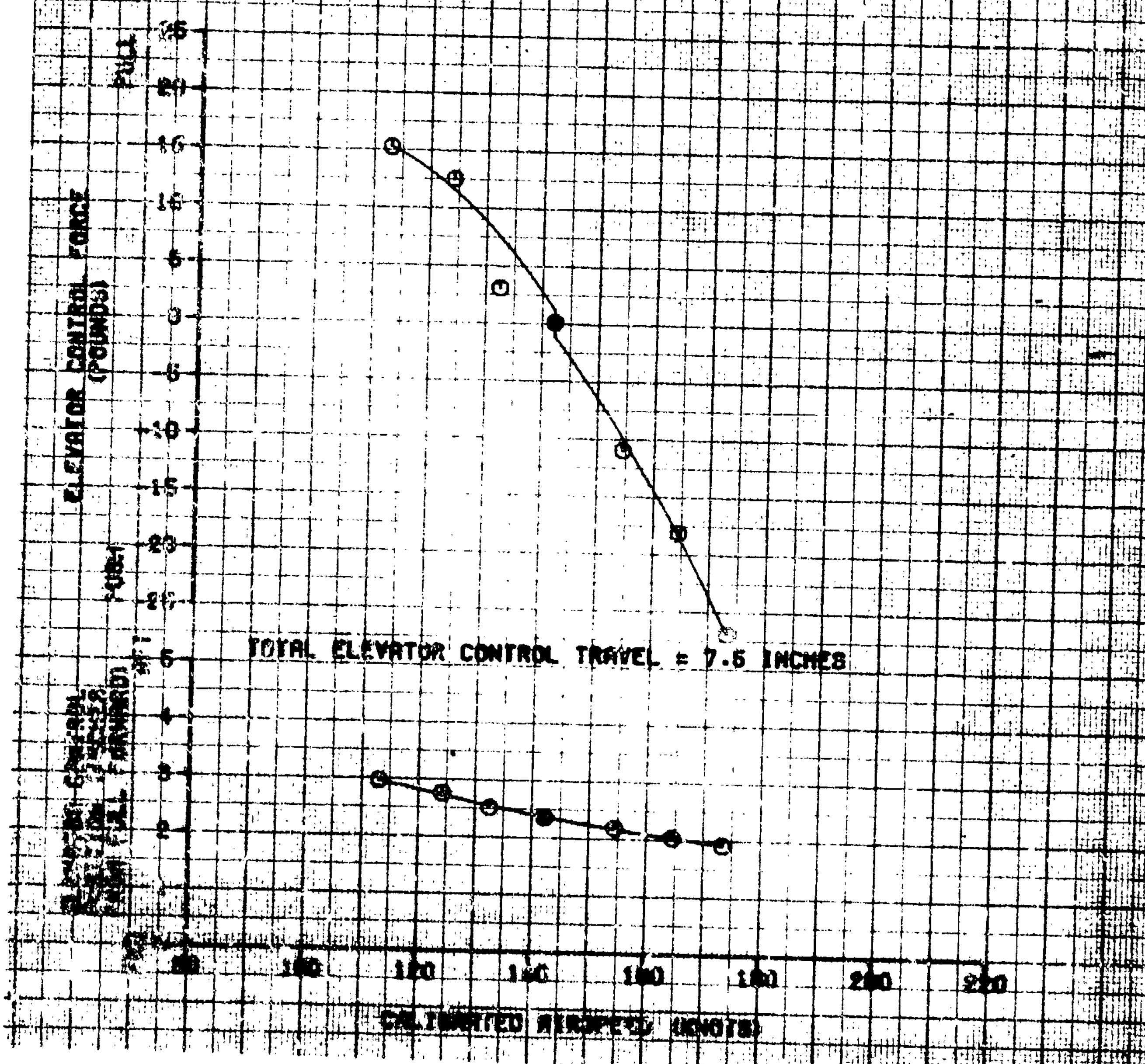


STATIC LONGITUDINAL STABILITY
AVG GROSS WEIGHT (LB)
AVG LONG CG LOCATION
AVG DENSITY ALTITUDE (FT)
AVG OAT (C)
PROPELLER SPEED (RPM)
CONFIGURATION
FLIGHT CONDITION
12040
11250
1800
TAKEOFF
CLIMB
NOTE: SHADED SYMBOLS DENOTE TRIM
ELEVATOR CONTROL FORCE (POUNDS)
PULL
PUSH
25
20
15
10
5
0
-5
-10
-15
-20
-25
TOTAL ELEVATOR CONTROL TRAVEL = 7.5 INCHES
ELEVATOR CONTROL POSITION (INCHES FROM FULL FORWARD)
AFT
FWD
6
4
2
1
0
80
100
120
140
160
180
200
220

FIGURE 66
STATIC LONGITUDINAL STABILITY
C-12A USA S/N 73-22250

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	PROPELLER SPEED (RPM)	CONFIGURATION	FLIGHT CONDITION
11300	185.1(FWD)	10050	-1.0	1815	TAKEOFF	LEVEL FLT

NOTE: SHADED SYMBOLS DENOTE TRIM



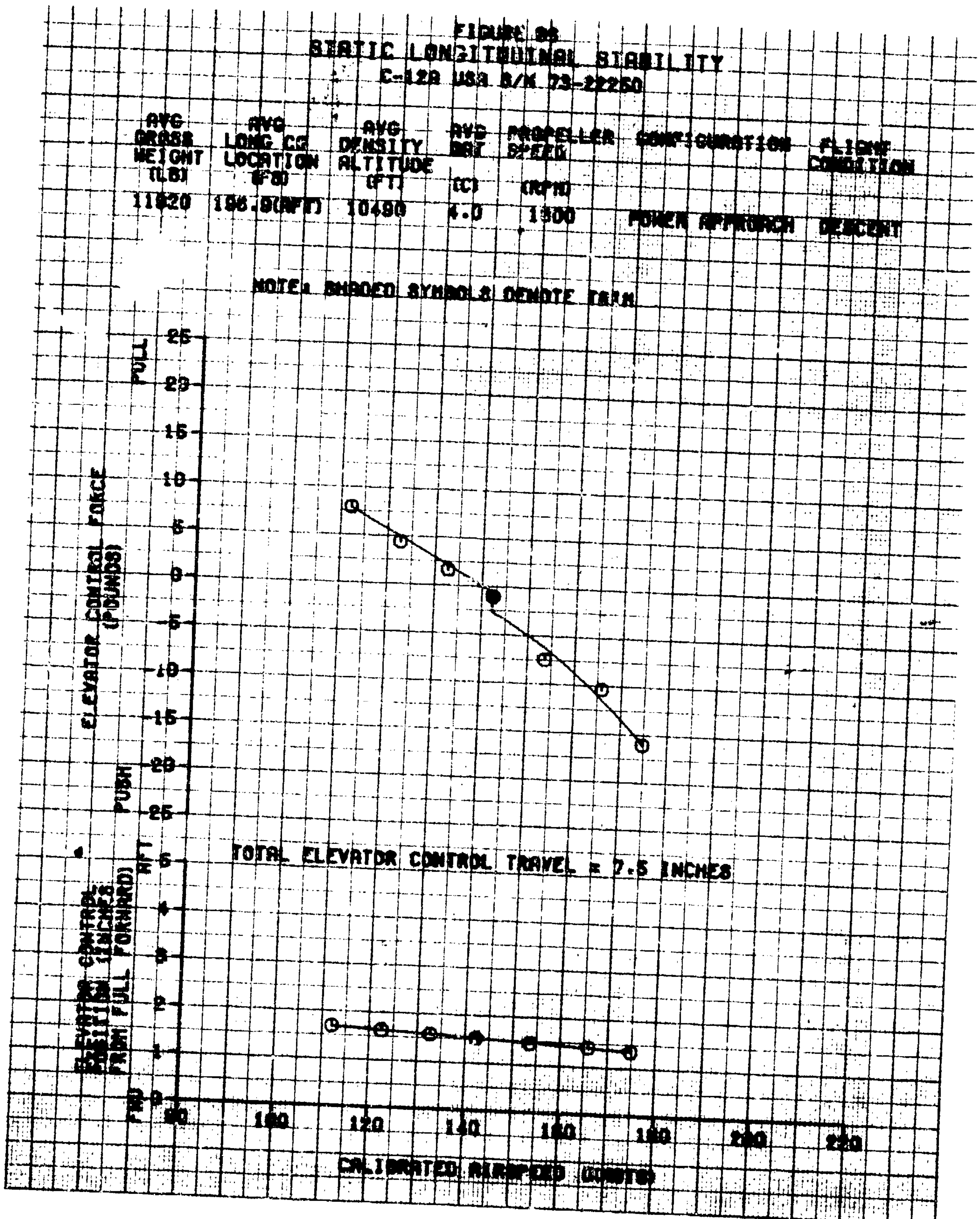
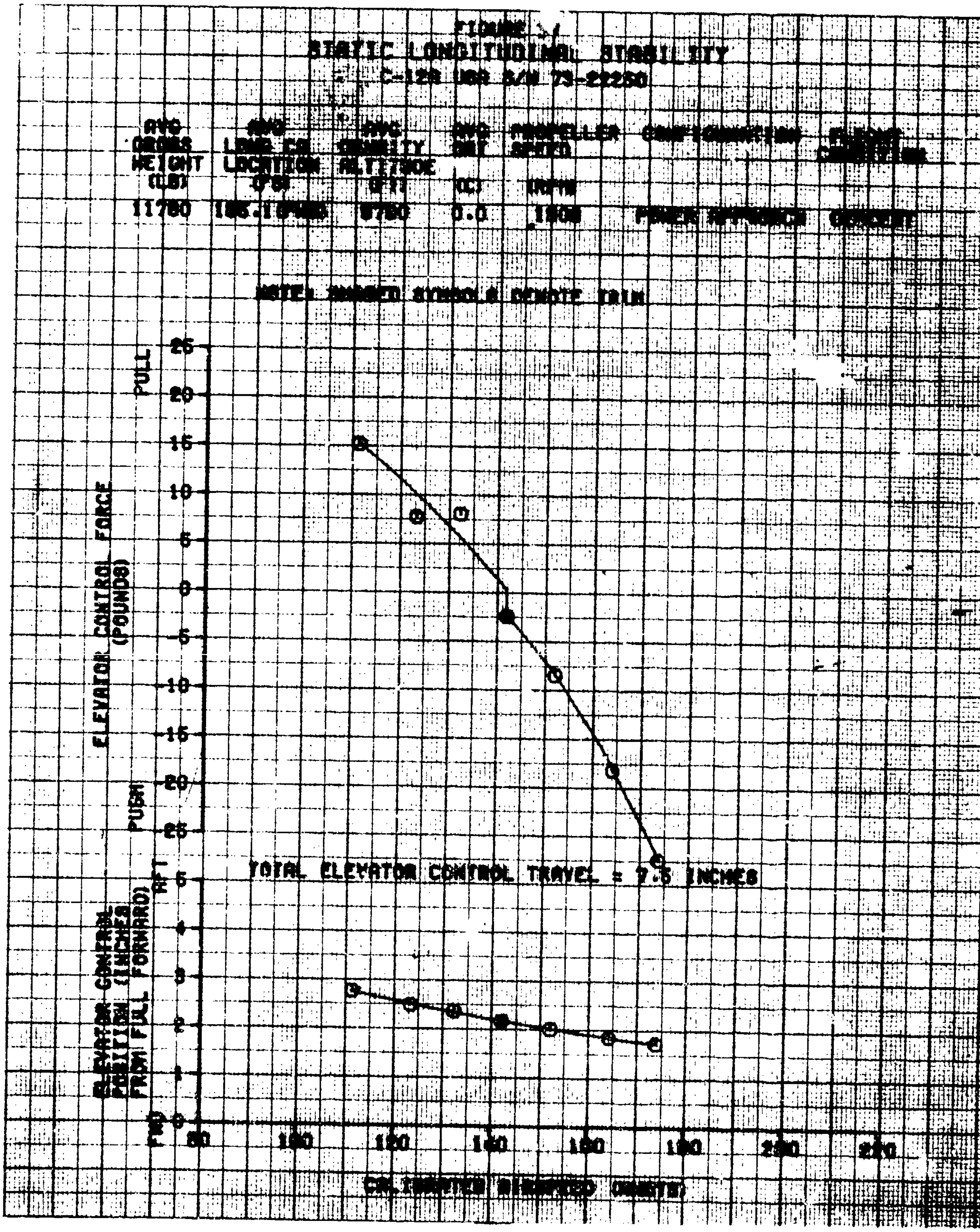


FIGURE 98
STATIC LONGITUDINAL STABILITY
C-12A USA S/N 73-22250
AVG GROSS WEIGHT (LB)
AVG LONG CG LOCATION (FS)
AVG DENSITY ALTITUDE (FT)
AVG OAT (C)
PROPELLER SPEED (RPM)
CONFIGURATION
FLIGHT CONDITION
11820
196.9(AFT)
10490
4.0
1900
POWER APPROACH
DESCENT
NOTE: SHADED SYMBOLS DENOTE TRIM
ELEVATOR CONTROL FORCE (POUNDS)
PULL
PUSH
TOTAL ELEVATOR CONTROL TRAVEL = 7.5 INCHES
ELEVATOR CONTROL POSITION (INCHES FROM FULL FORWARD)
AFT
FWD
CALIBRATED AIRSPEED (KNOTS)



STATIC LONGITUDINAL STABILITY
11780
0.0
ELEVATOR CONTROL FORCE (POUNDS)
PULL
PUSH
25
20
15
10
5
0
-5
-10
-15
-20
-25
AFT
FWD
6
4
3
2
1
0
80
100
120
140
160
180
200
220

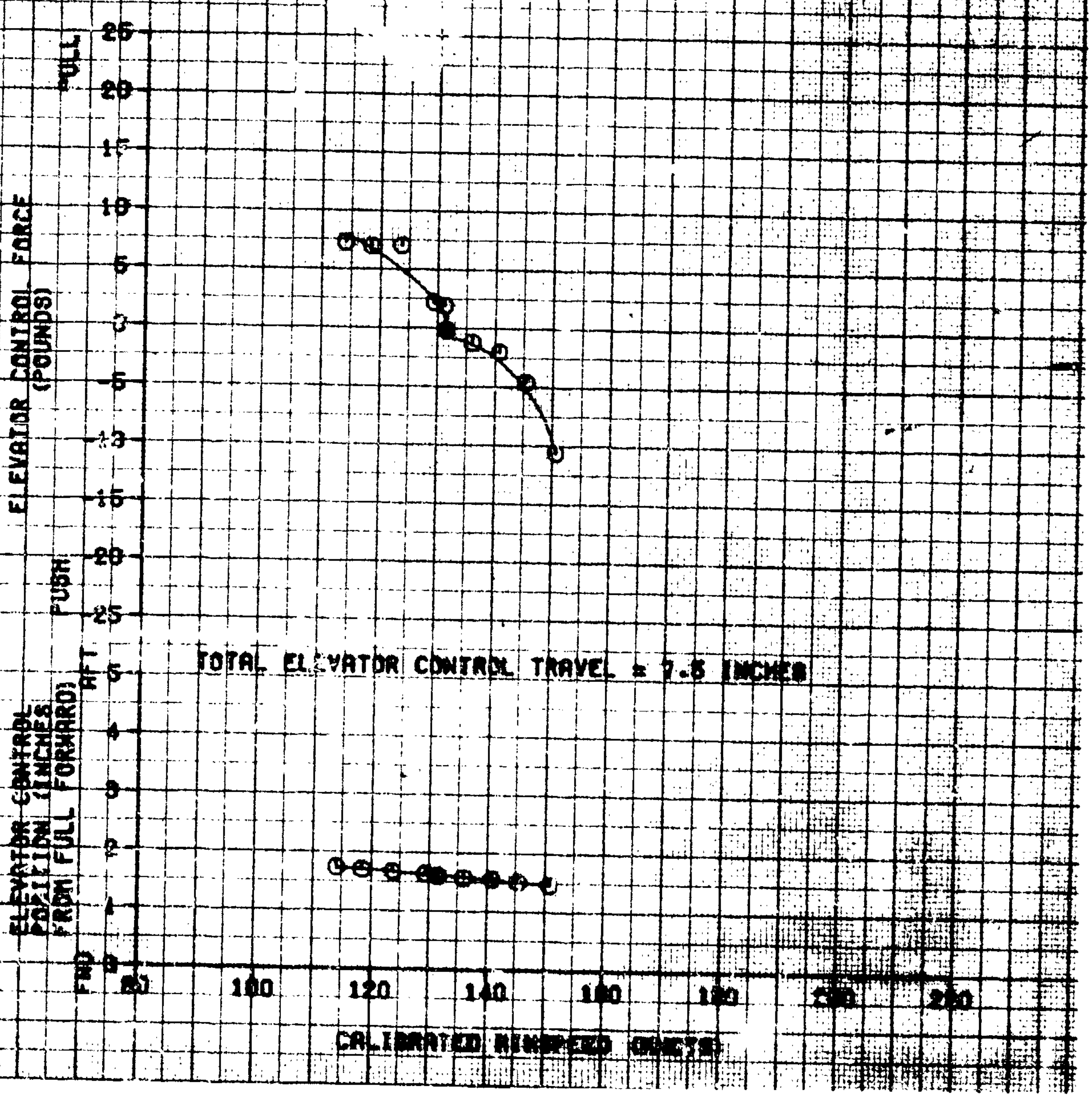
FIGURE 88

STATIC LONGITUDINAL STABILITY

C-12A USA S/N 73-22250

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	PROPELLER SPEED (RPM)	CONFIGURATION	FLIGHT CONDITION
12580	158.2(AFT)	11470	5.0	1800	POWER APPROACH	DESCENT

NOTE: SHADED SYMBOLS DENOTE TRIM



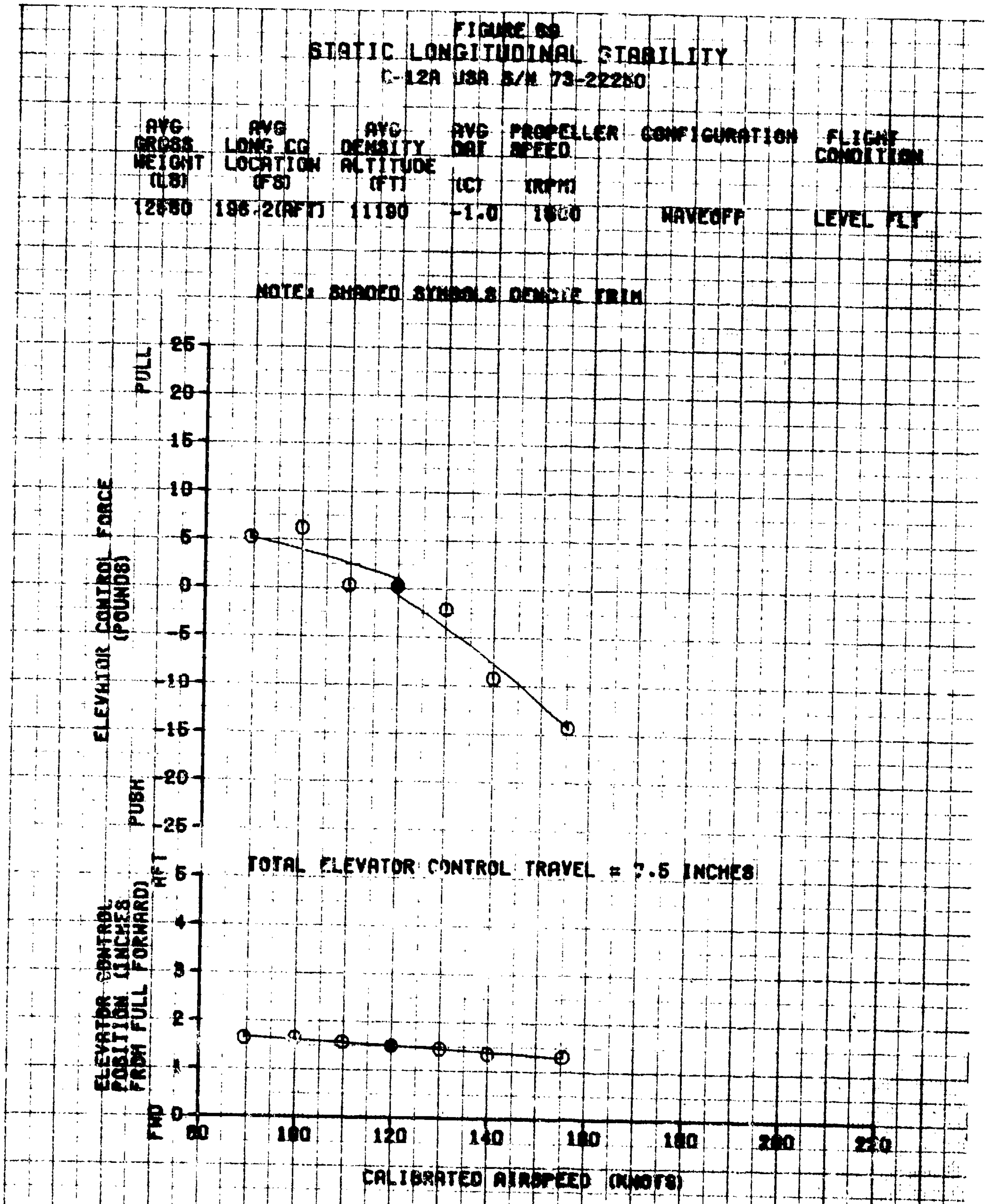
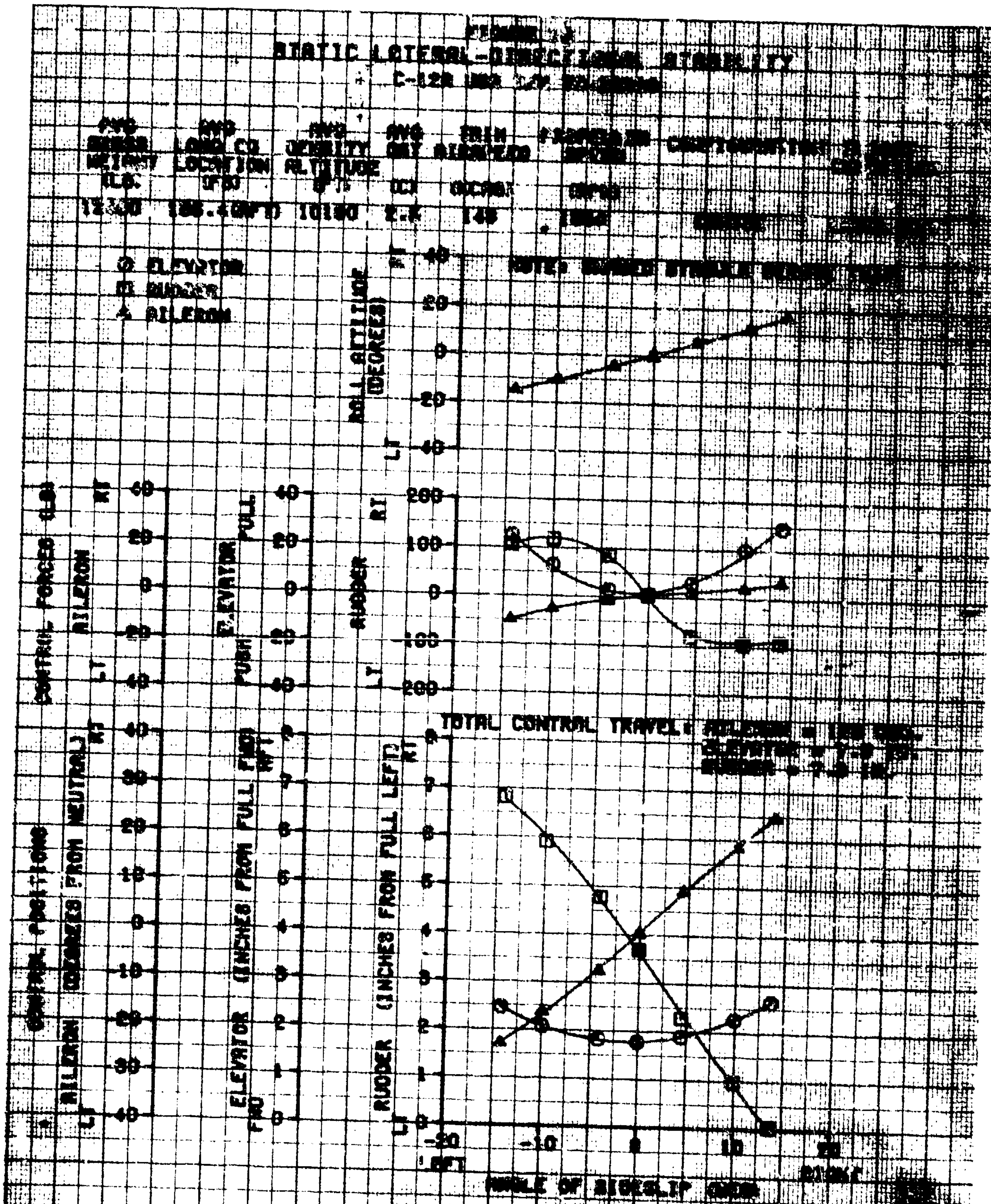


FIGURE 89
STATIC LONGITUDINAL STABILITY
C-12A USA S/N 73-22250
AVG GROSS WEIGHT (LB) | AVG LONG CG LOCATION (FS) | AVG DENSITY ALTITUDE (FT) | AVG OAT (C) | PROPELLER SPEED (RPM) | CONFIGURATION | FLIGHT CONDITION
12550 | 196.2(AFT) | 11190 | -1.0 | 1800 | WAVEOFF | LEVEL FLT
NOTE: SHADED SYMBOLS DENOTE TRIM
ELEVATOR CONTROL FORCE (POUNDS)
PULL
PUSH
TOTAL ELEVATOR CONTROL TRAVEL = 7.5 INCHES
ELEVATOR CONTROL POSITION (INCHES FROM FULL FORWARD)
AFT
FWD
CALIBRATED AIRSPEED (KNOTS)


STATIC LATERAL-DIRECTIONAL STABILITY
ELEVATOR
RUDDER
AILERON
ROLL ATTITUDE (DEGREES)
CONTROL FORCES (LB)
CONTROL POSITIONS
TOTAL CONTROL TRAVEL:
ANGLE OF SIDESLIP

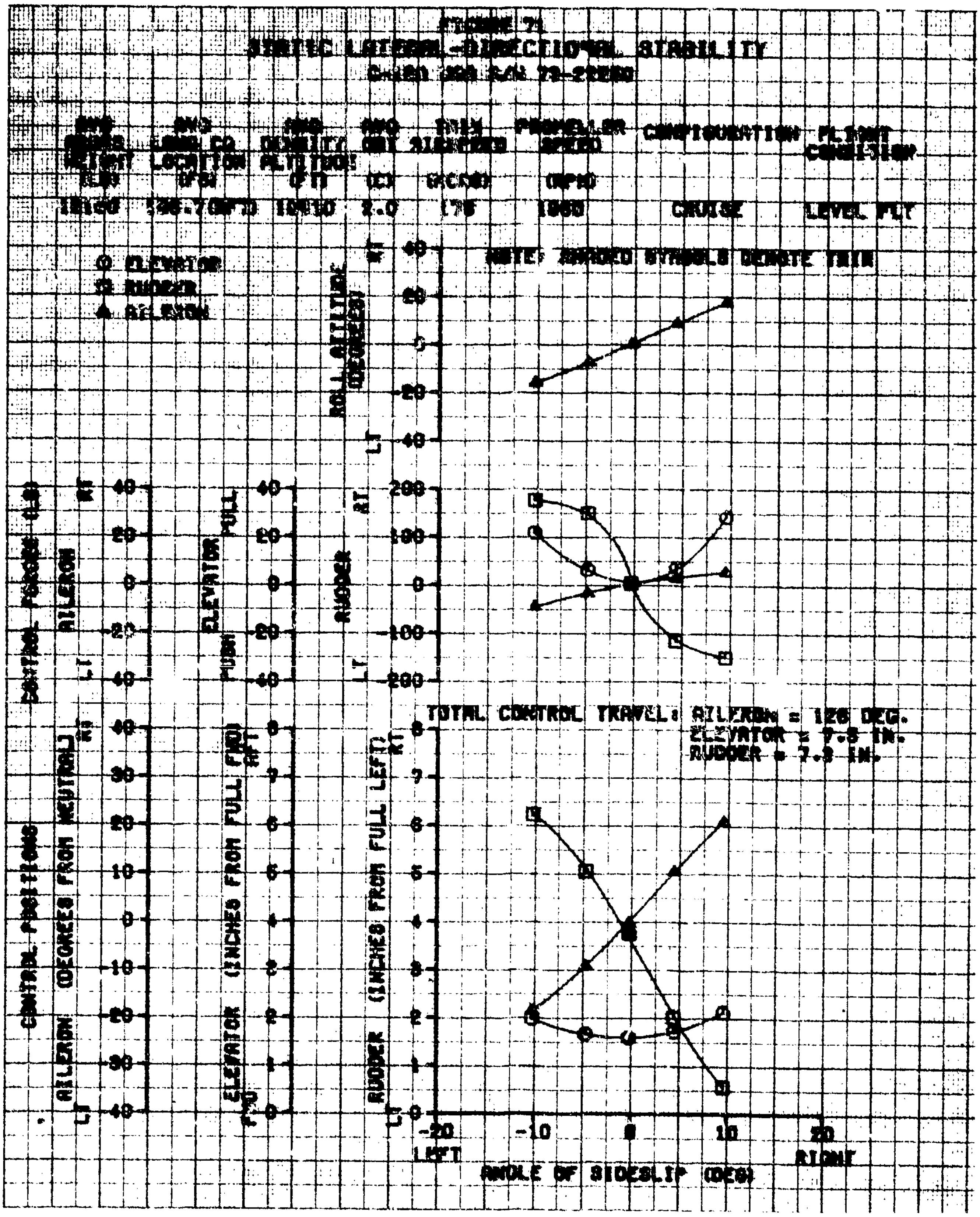
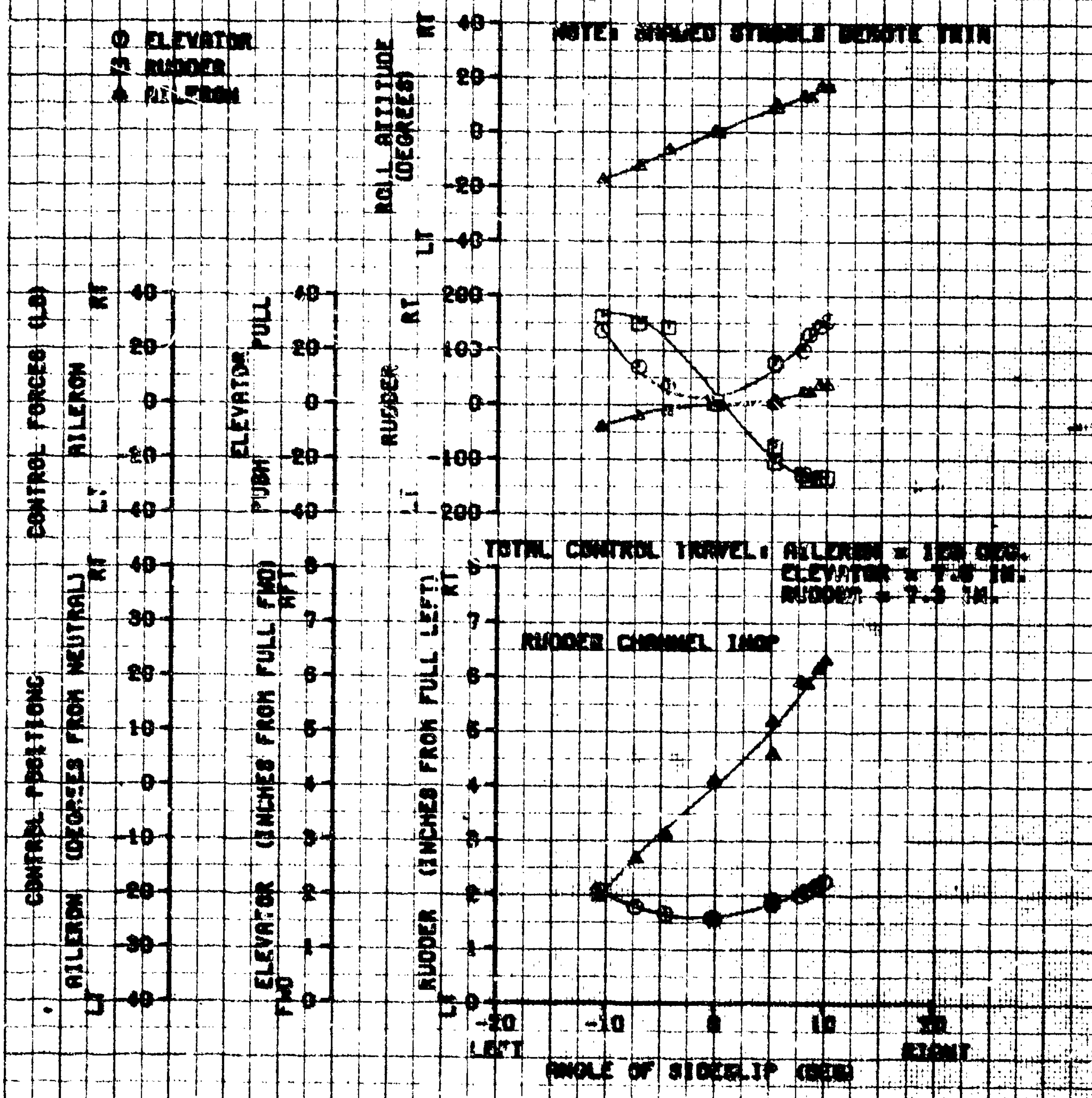

CRUISE
LEVEL FLT
ELEVATOR
RUDDER
AILERON
ROLL ATTITUDE (DEGREES)
CONTROL FORCES (LB)
AILERON
ELEVATOR
RUDDER
PULL
PUSH
CONTROL POSITIONS
AILERON (DEGREES FROM NEUTRAL)
ELEVATOR (INCHES FROM FULL FWD)
RUDDER (INCHES FROM FULL LEFT)
TOTAL CONTROL TRAVEL:
LEFT
RIGHT
ANGLE OF SIDESLIP (DEG)

FIGURE 72
STATIC LATERAL-DIRECTIONAL STABILITY
C-12A USA S/N 73-22250

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	TRIM AIRSPEED (KCAS)	PROPELLER SPEED (RPM)	CONFIGURATION	FLIGHT CONDITION
11840	186.8(AFT)	24080	-27.5	185	1800	CRUISE	LEVEL FLT



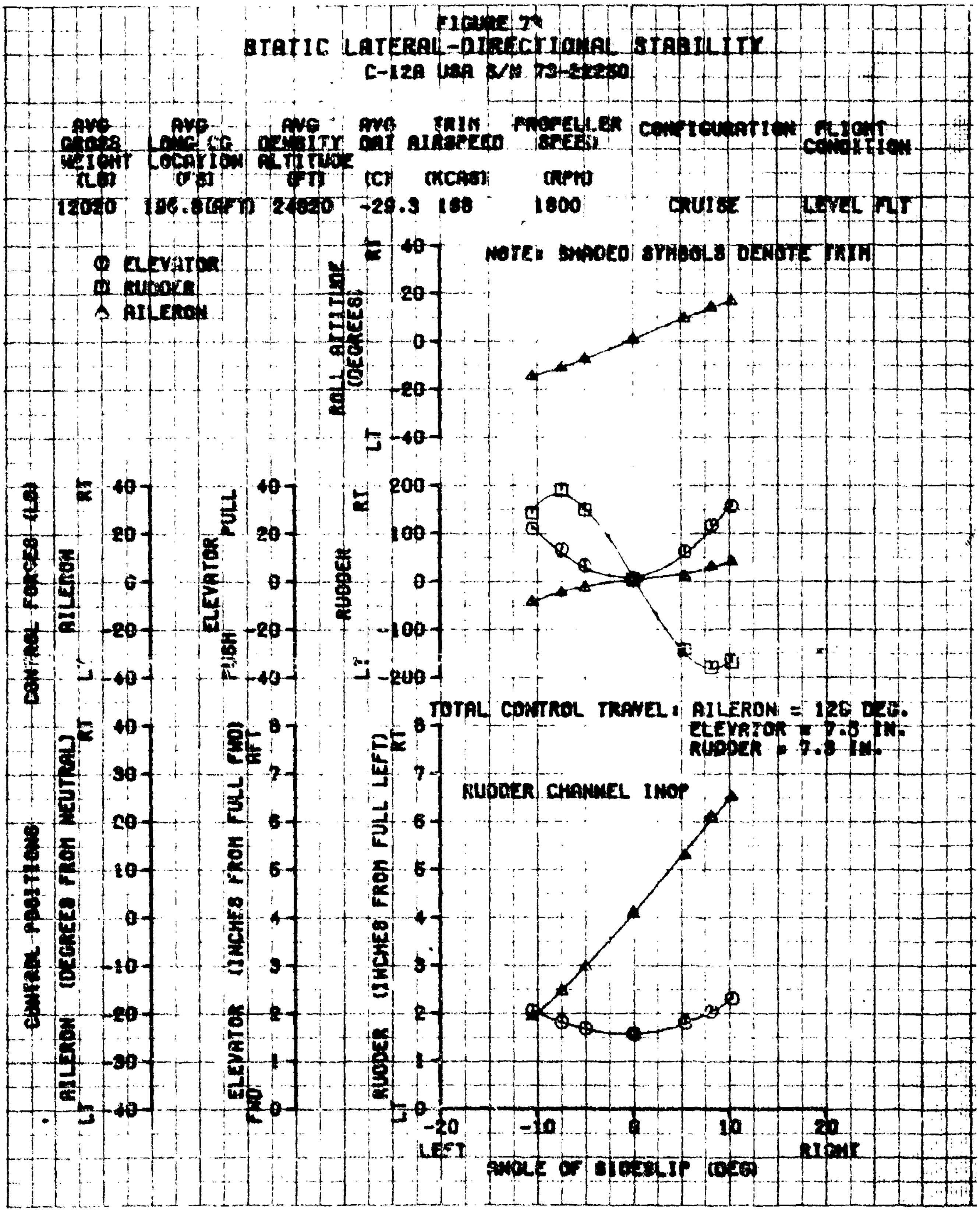


FIGURE 7
STATIC LATERAL-DIRECTIONAL STABILITY
C-12A USA S/N 73-22250
AVG GROSS WEIGHT (LB) 12020
AVG LONG CG LOCATION (FS) 196.8(AFT)
AVG DENSITY ALTITUDE (FT) 24820
AVG OAT (C) -29.3
TRIM AIRSPEED (KCAS) 168
PROPELLER SPEED (RPM) 1800
CONFIGURATION CRUISE
FLIGHT CONDITION LEVEL FLT
ELEVATOR
RUDDER
AILERON
NOTE: SHADED SYMBOLS DENOTE TRIM
ROLL ATTITUDE (DEGREES)
CONTROL FORCES (LB)
CONTROL POSITIONS
TOTAL CONTROL TRAVEL: AILERON = 126 DEG.
ELEVATOR = 7.8 IN.
RUDDER = 7.8 IN.
RUDDER CHANNEL INOP
AILERON (DEGREES FROM NEUTRAL)
ELEVATOR (INCHES FROM FULL FWD)
RUDDER (INCHES FROM FULL LEFT)
ANGLE OF SIDESLIP (DEG)
LEFT
RIGHT

FIGURE 74
STATIC LATERAL-DIRECTIONAL STABILITY
C-12A USA S/N 73-22250

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	TRIM AIRSPEED (KCAS)	PROPELLER SPEED (RPM)	CONFIGURATION	FLIGHT CONDITION
12840	196.2(AFT)	10540	2.0	121	1650	POWER APPROACH	DESCENT

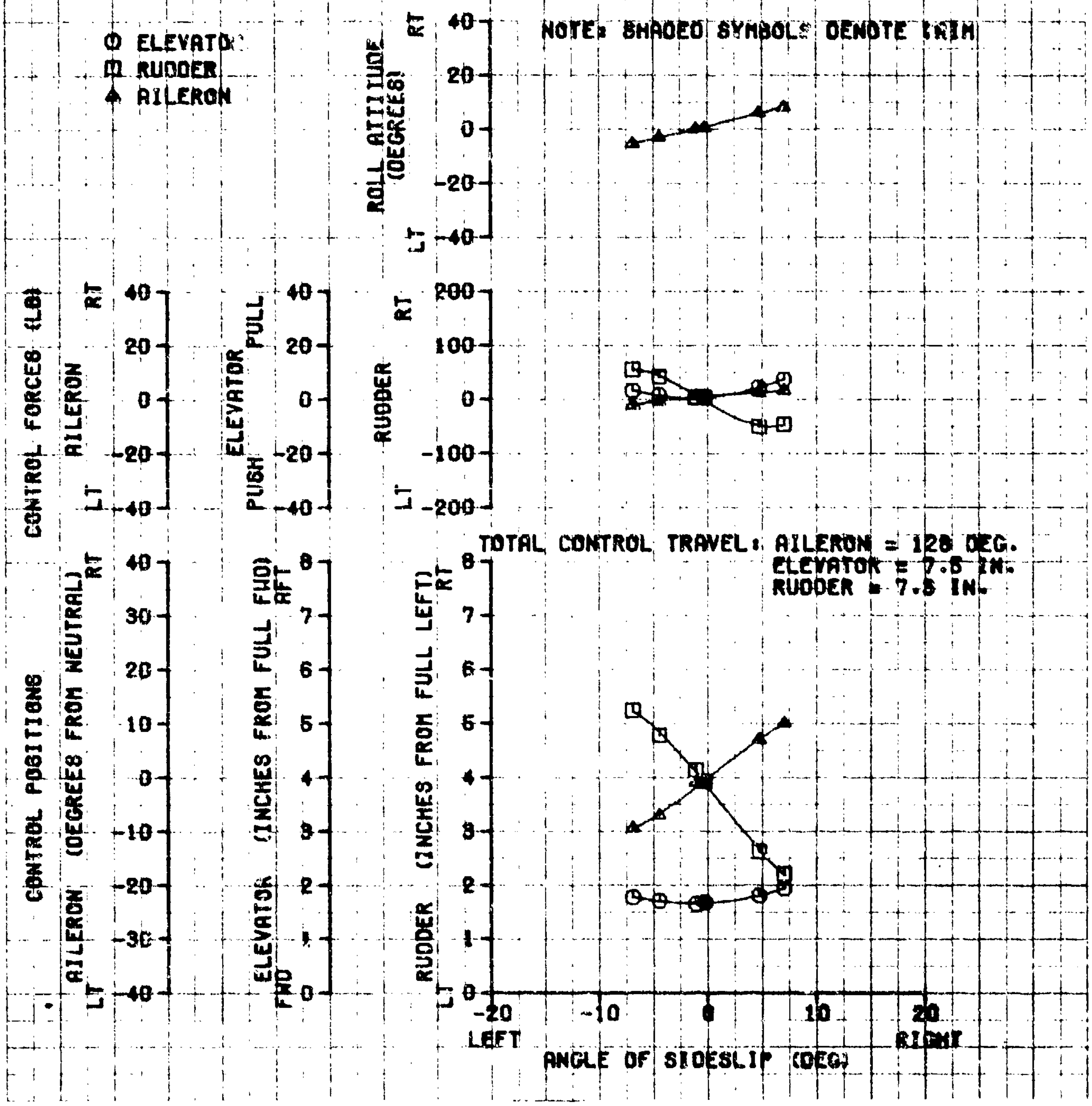


FIGURE 75
DYNAMIC LONGITUDINAL STABILITY (PHUGOID)
C-12A USA S/N 73-22250

TRIM AIRSPEED (KCAS)	AVG GROSS WEIGHT (LB)	AVG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	CONFIGURATION	FLIGHT CONDITION
189	11610	197.1	10490	2.3	CRUISE	LEVEL FLIGHT
209	11700	197.0	10640	4.8	CRUISE	LEVEL FLIGHT
173	11480	197.2	11410	4.8	POWER APPROACH	LEVEL FLIGHT

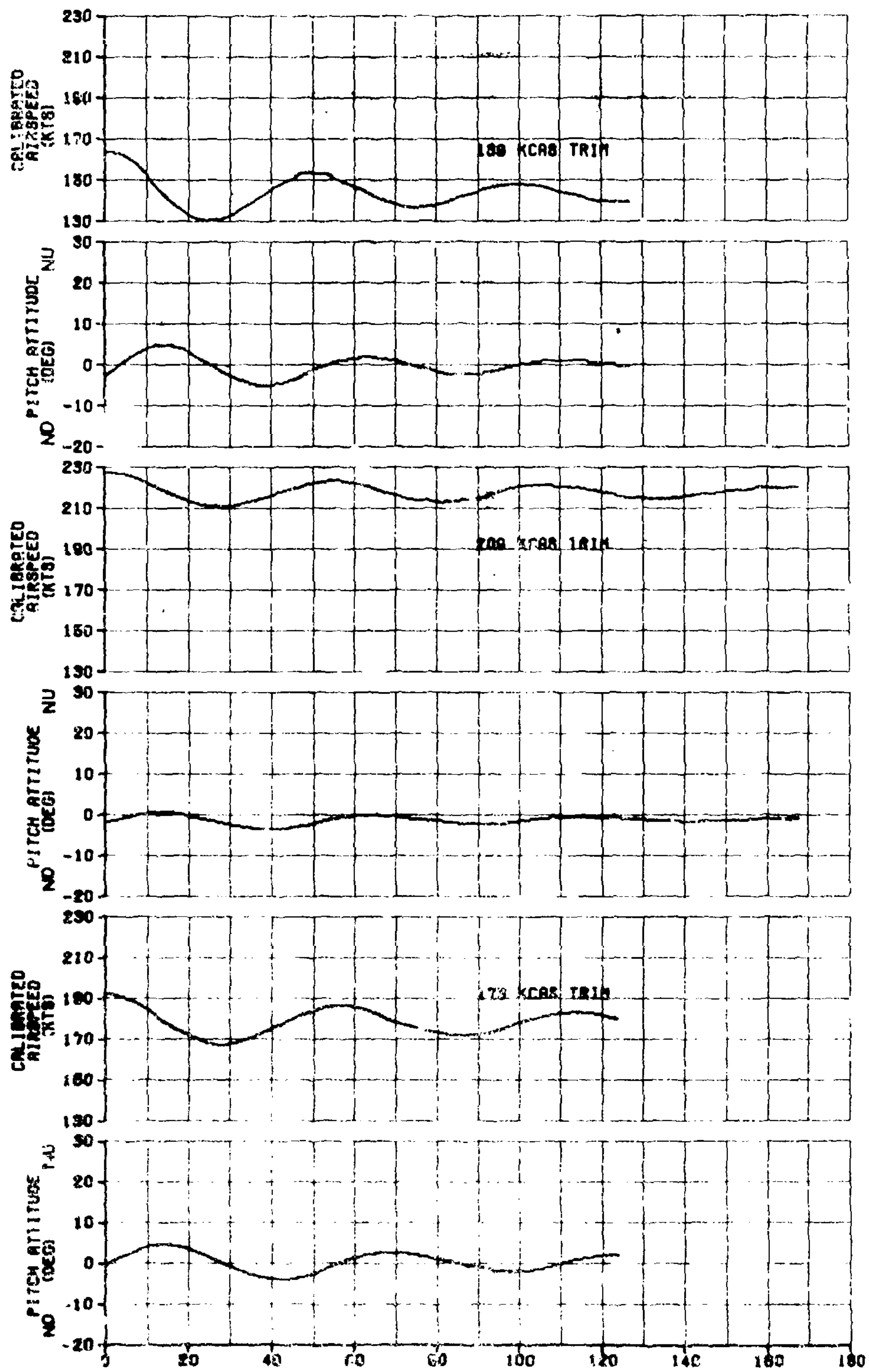


FIGURE 76
DYNAMIC LONGITUDINAL STABILITY (SHORT PERIOD RESPONSE)
C-12A USA S/N 73-22260

AVG GROSS WEIGHT (LB)	AVG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	CONFIGURATION	FLIGHT CONDITION
11840	186.1(FWD)	16820	-0.5	CRUISE	LEVEL FLIGHT

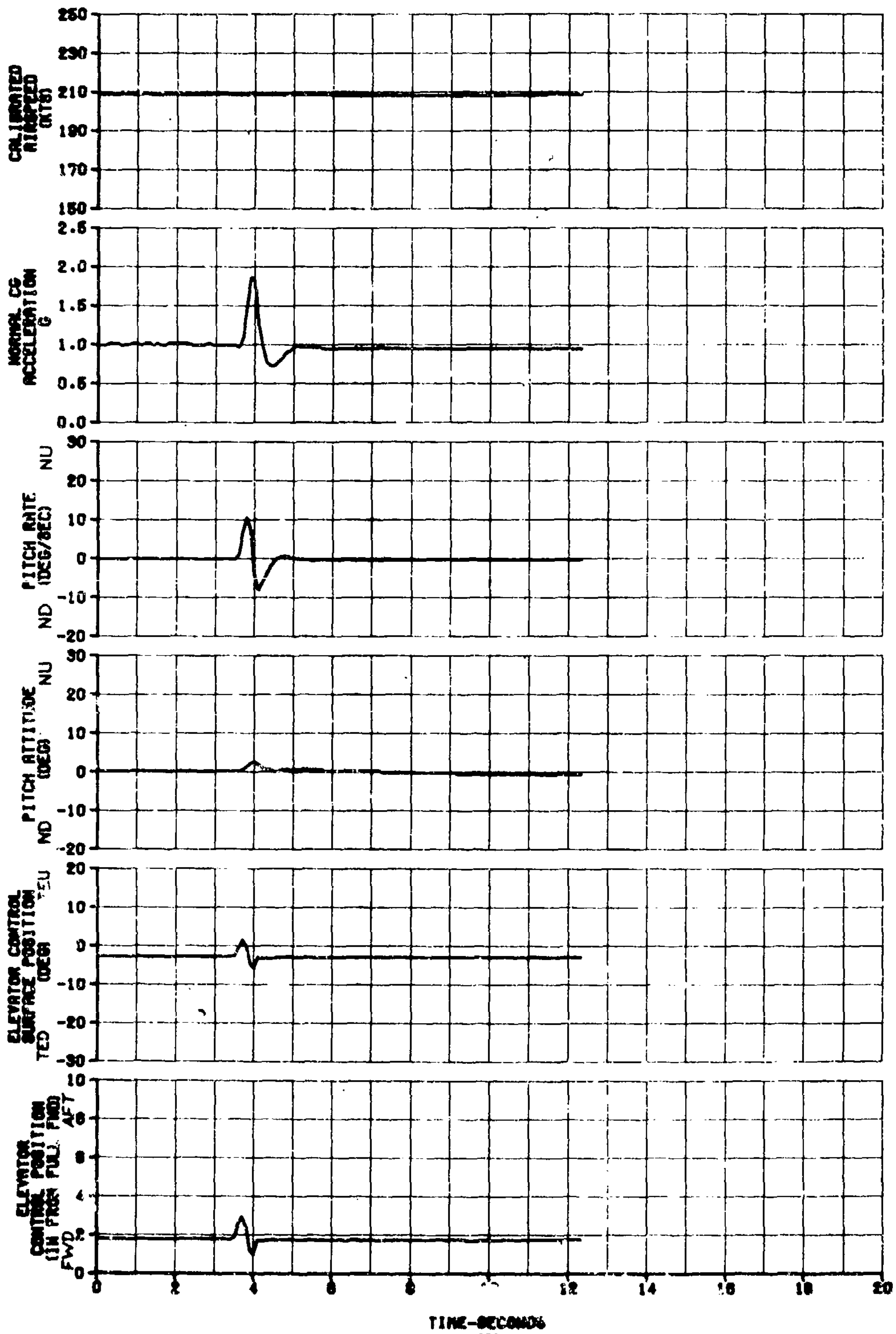


FIGURE 77
AIRCRAFT RESPONSE DURING SIMULATED CROSSWIND LANDING
C-12H USA S/N 73-22250

GROSS WEIGHT (LB)	LONG CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (C)	PROPELLER SPEED (RPM)	TRIM AIRSPEED (KCAS)	CONFIGURATION	FLIGHT CONDITION
12570	196.2 (AFT)	10520	2.0	1900	131	POWER APPROACH	STDY HEADING SIDESLIP

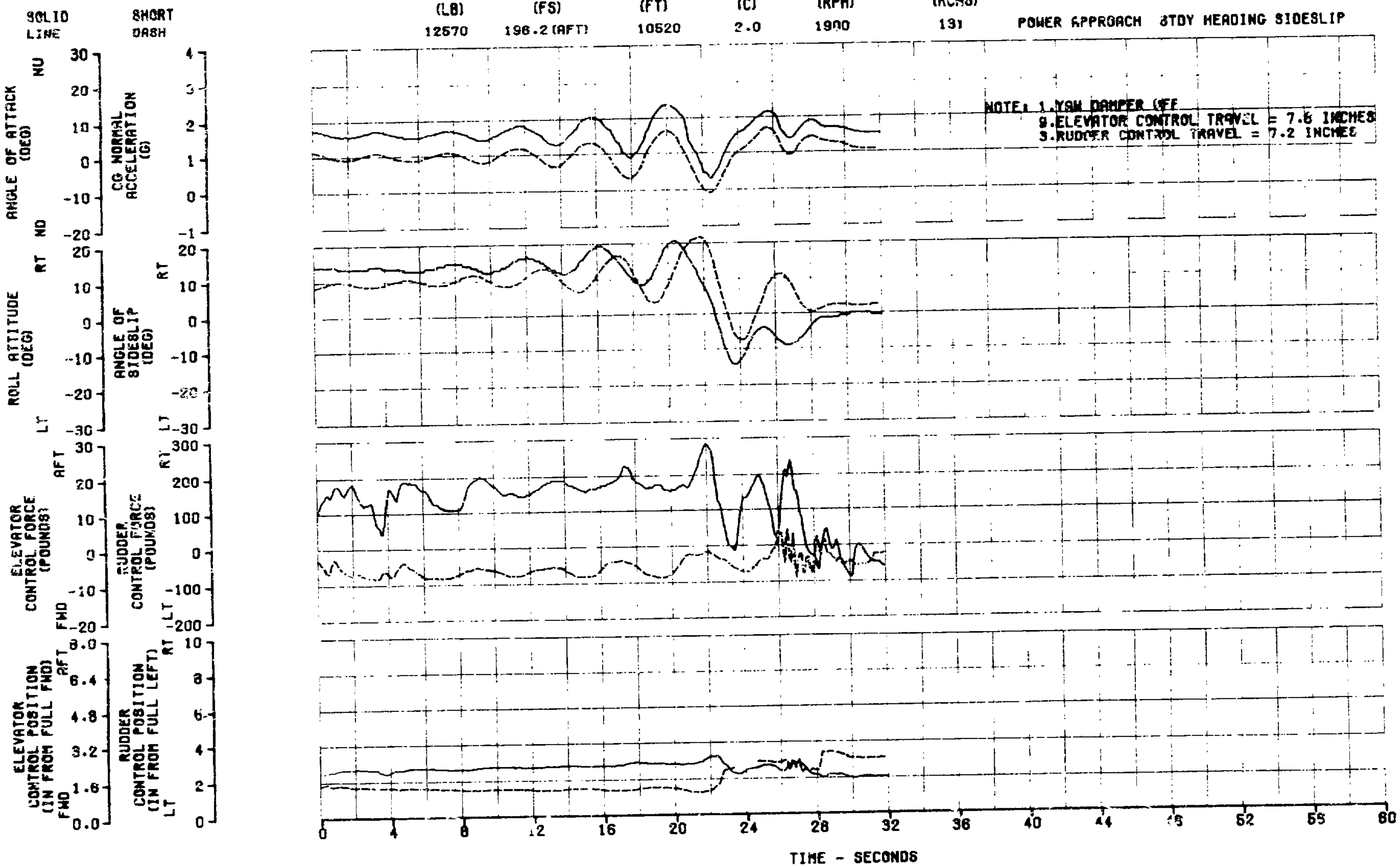


FIGURE 78
DUTCH ROLL
C-12A USA S/N 73-22250

GROSS WEIGHT (LB)	LONG CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (C)	PROPELLER SPEED (RPM)	TRIM AIRSPEED (KCAS)	CONFIGURATION	FLIGHT CONDITION
11490	197.2 (AFT)	25390	-3.0	1600	171	CRUISE	LEVEL FLIGHT

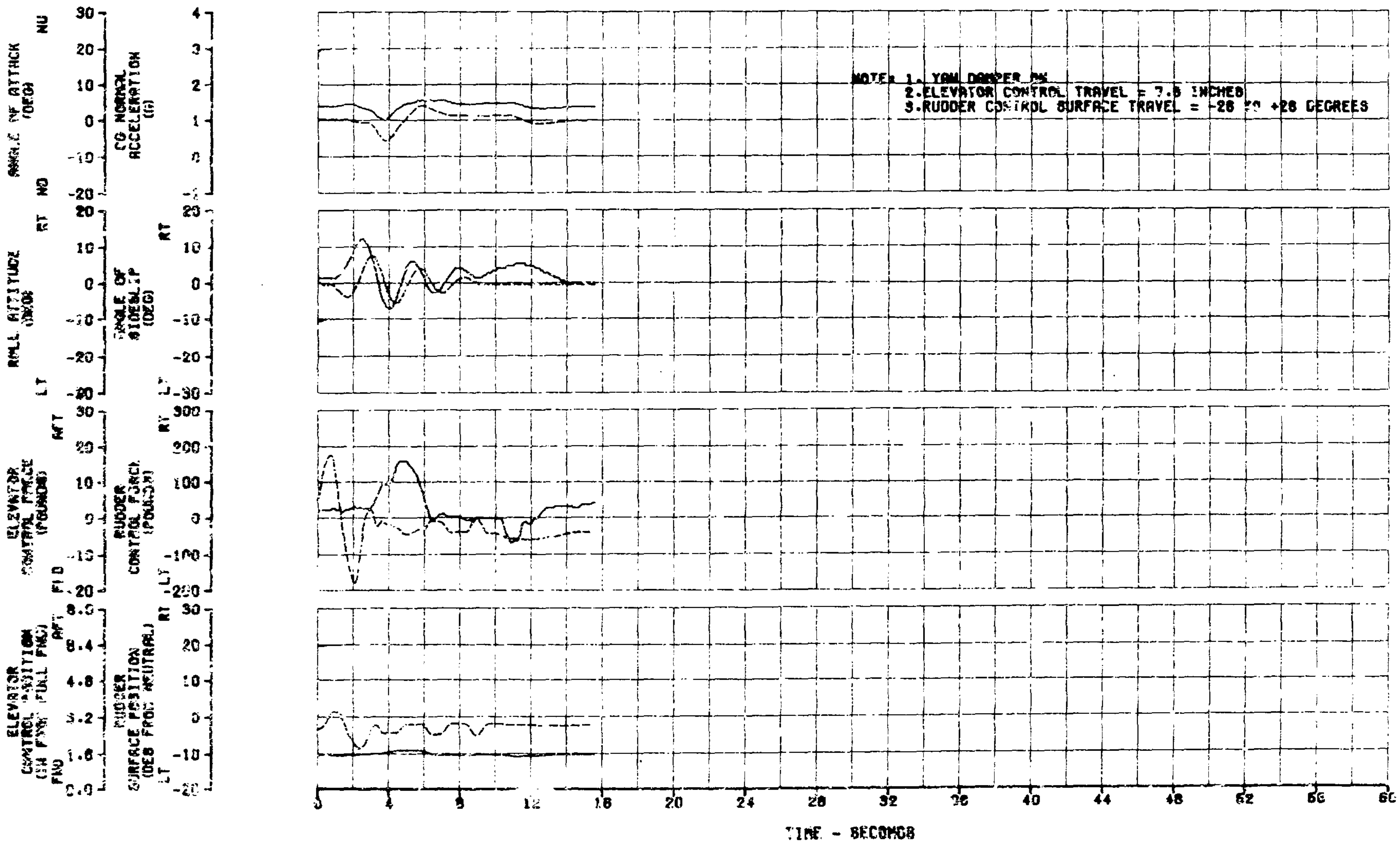


FIGURE 79
DUTCH ROLL
C-12A USA S/N 73-22250

GROSS WEIGHT (LB)	LONG CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (C)	PROPELLER SPEED (RPM)	TRIM AIRSPEED (KCAS)	CONFIGURATION	FLIGHT CONDITION
11650	187.2 (AFT)	25000	-3.0	1800	172	CRUISE	LEVEL FLIGHT

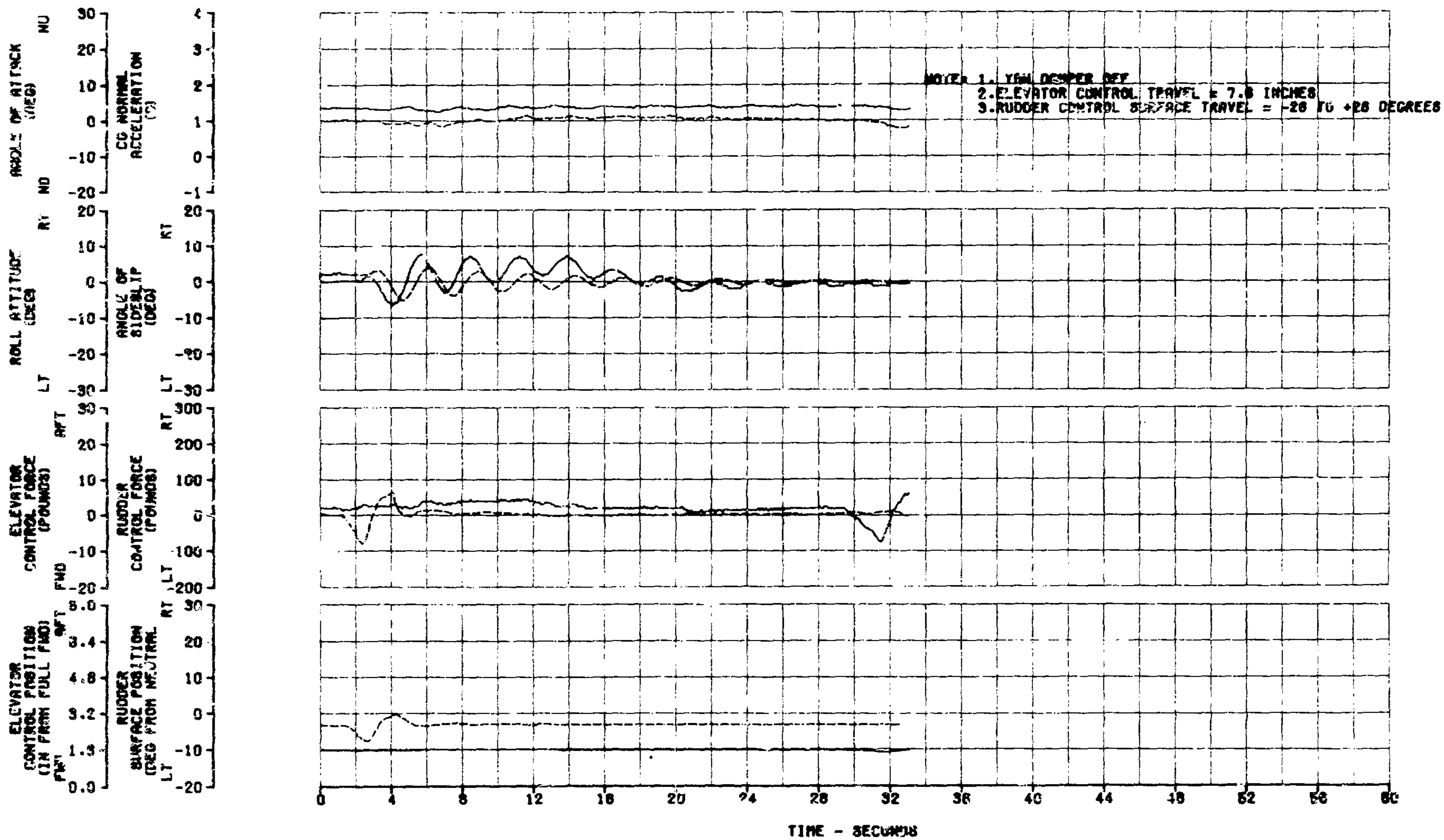


FIGURE 80
DUTCH ROLL
C-12A USA S/N 73-22250

GROSS WEIGHT (LB)	LONG CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (C)	PROPELLER SPEED (RPM)	TRIM AIRSPEED (KCAS)	CONFIGURATION	FLIGHT CONDITION
12410	186.5 (AFT)	12050	-5.5	1800	172	CRUISE	DESCENT

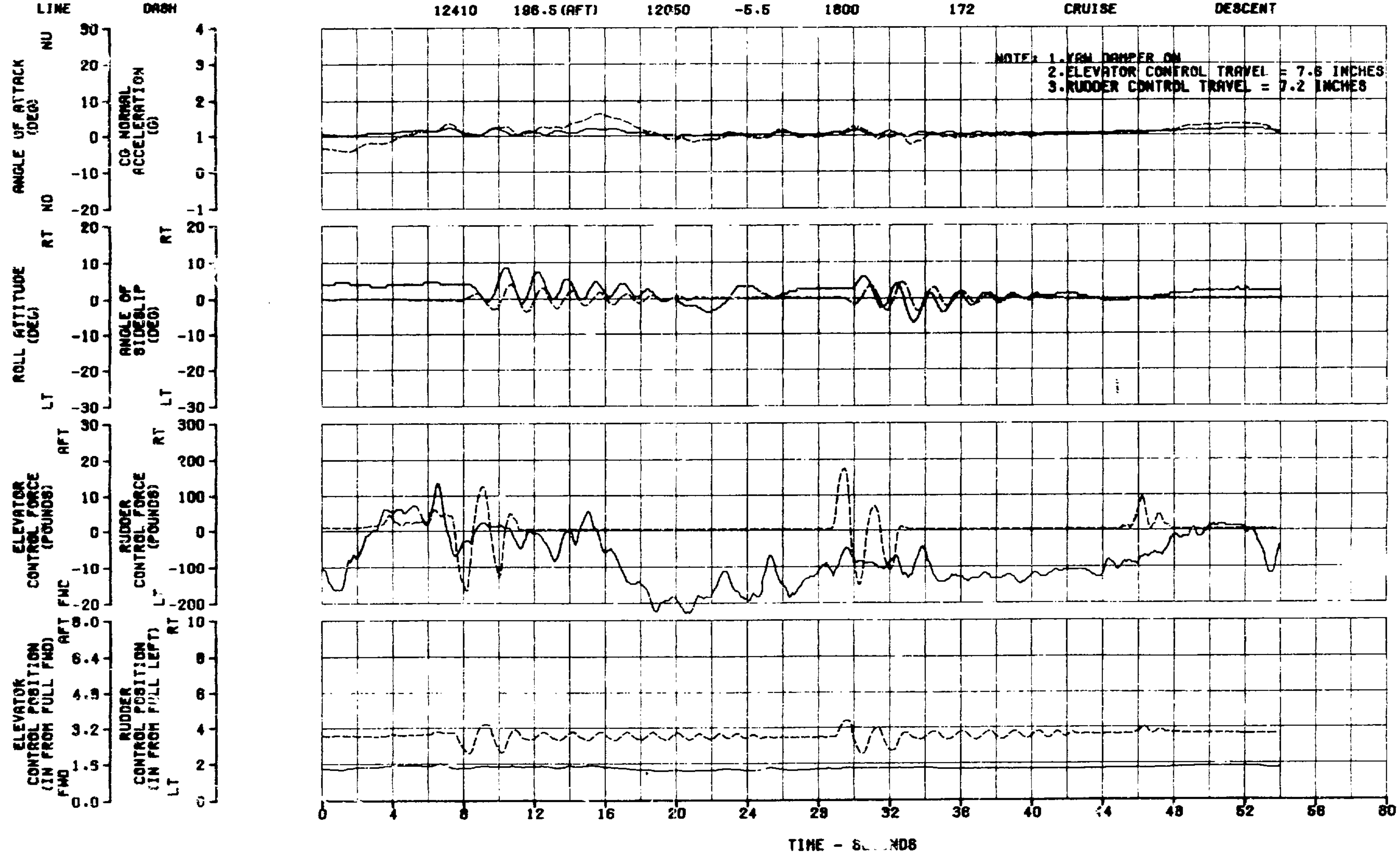


FIGURE 81
DUTCH ROLL
C-12A USA S/N 73-22250

GROSS WEIGHT (LB)	LONG CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (C)	PROPELLER SPEED (RPM)	TRIM AIRSPEED (KCAS)	CONFIGURATION	FLIGHT CONDITION
12500	196.4 (AFT)	9980	-2.5	1800	181	CRUISE	LEVEL FLIGHT

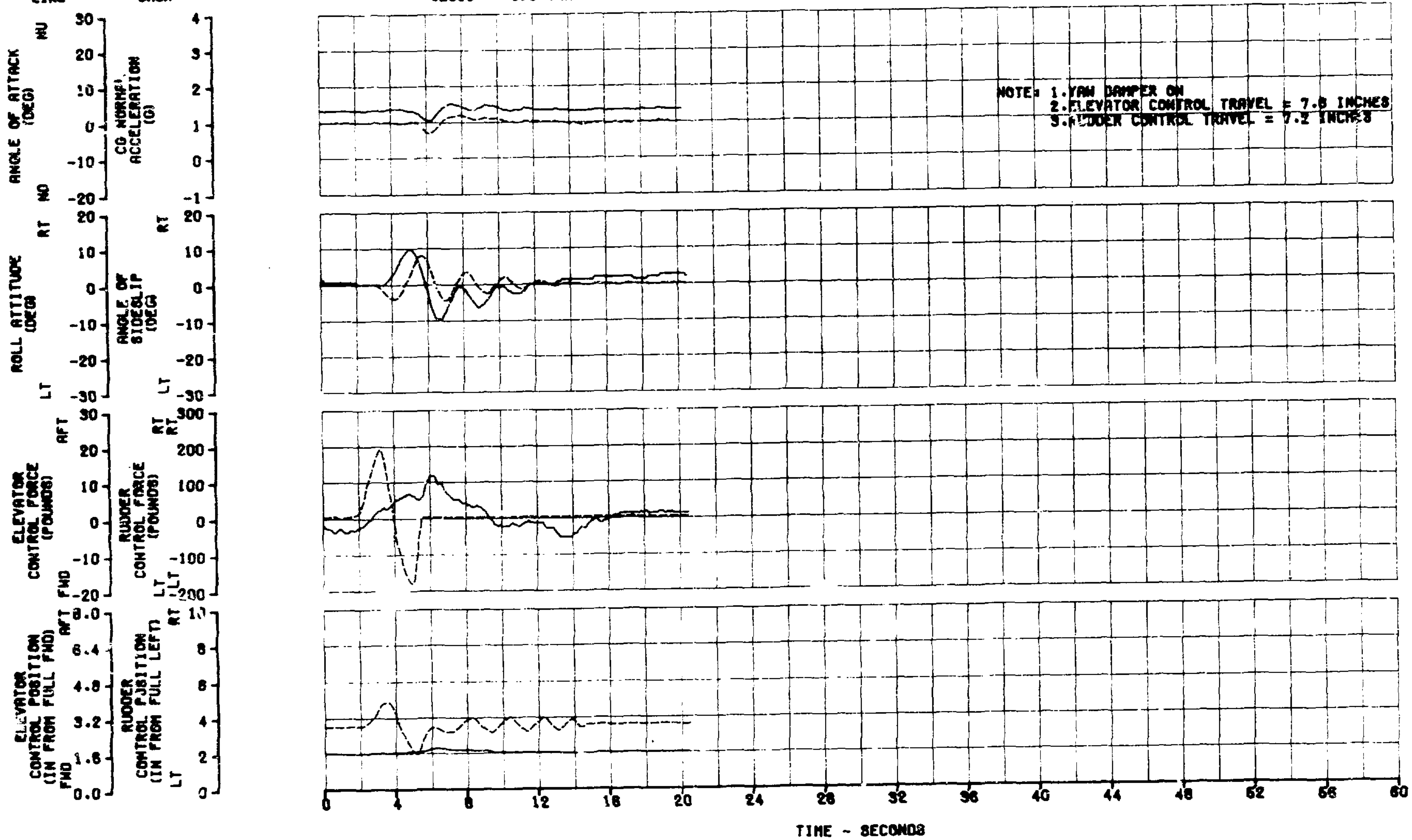


FIGURE 82
DUTCH ROLL
C-12A USA S/N 73-22250

GROSS WEIGHT (LB)	LONG CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (C)	PROPELLER SPEED (RPM)	TRIM AIRSPEED (KCAS)	CONFIGURATION	FLIGHT CONDITION
12540	196.4 (AFT)	10220	-3.0	1800	180	CRUISE	LEVEL FLIGHT

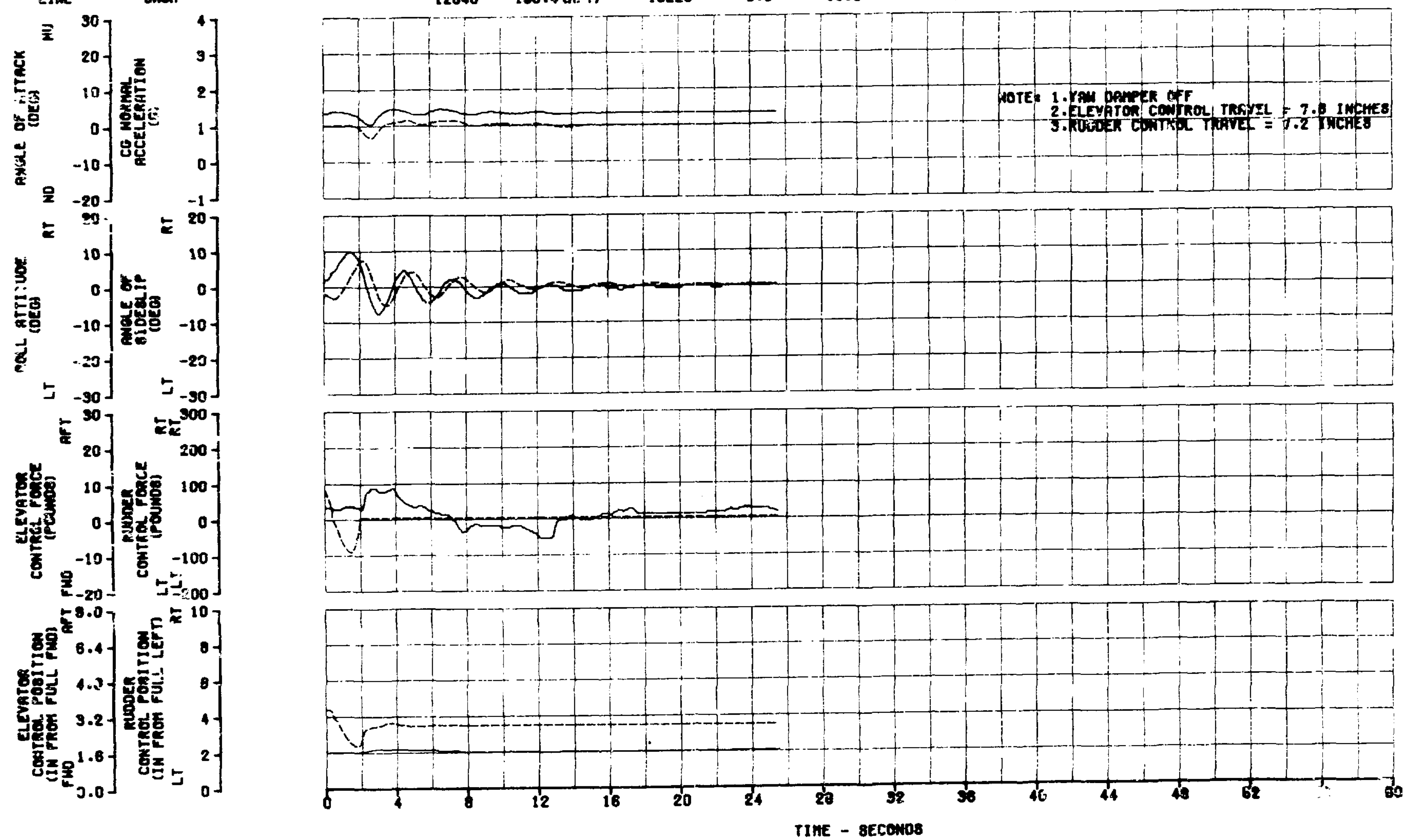


FIGURE 23
DUTCH ROLL
C-12A USA S/N 73-22250

GROSS WEIGHT (LB)	LONG CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (C)	PROPELLER SPEED (RPM)	TRIM AIRSPEED (KCAS)	CONFIGURATION	FLIGHT CONDITION
12730	196.2 (AFT)	10180	-2.0	1900	120	POWER APPROACH	LEVEL FLIGHT

FIGURE 84
DUTCH ROLL
C-12A USA S/N 73-22250

GROSS WEIGHT (LB)	LONG CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (C)	PROPELLER SPEED (RPM)	TRIM AIRSPEED (KCAS)	CONFIGURATION	FLIGHT CONDITION
12740	196.2 (AFT)	10330	-2.0	1900	120	POWER APPROACH	LEVEL FLIGHT

NOTES: 1. YAW DAMPER OFF
2. ELEVATOR CONTROL TRAVEL = 7.8 INCHES
3. RUDDER CONTROL TRAVEL = 7.2 INCHES

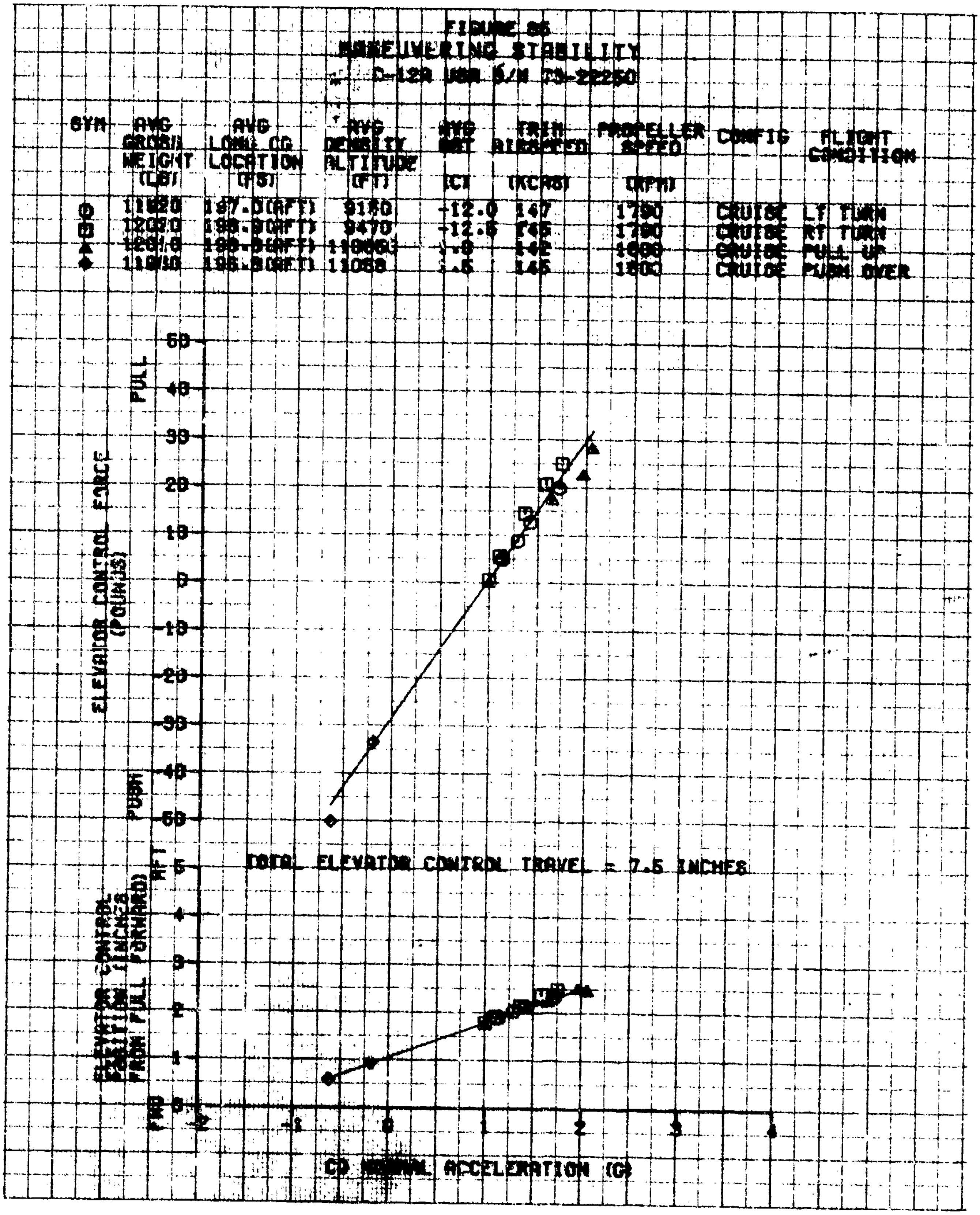
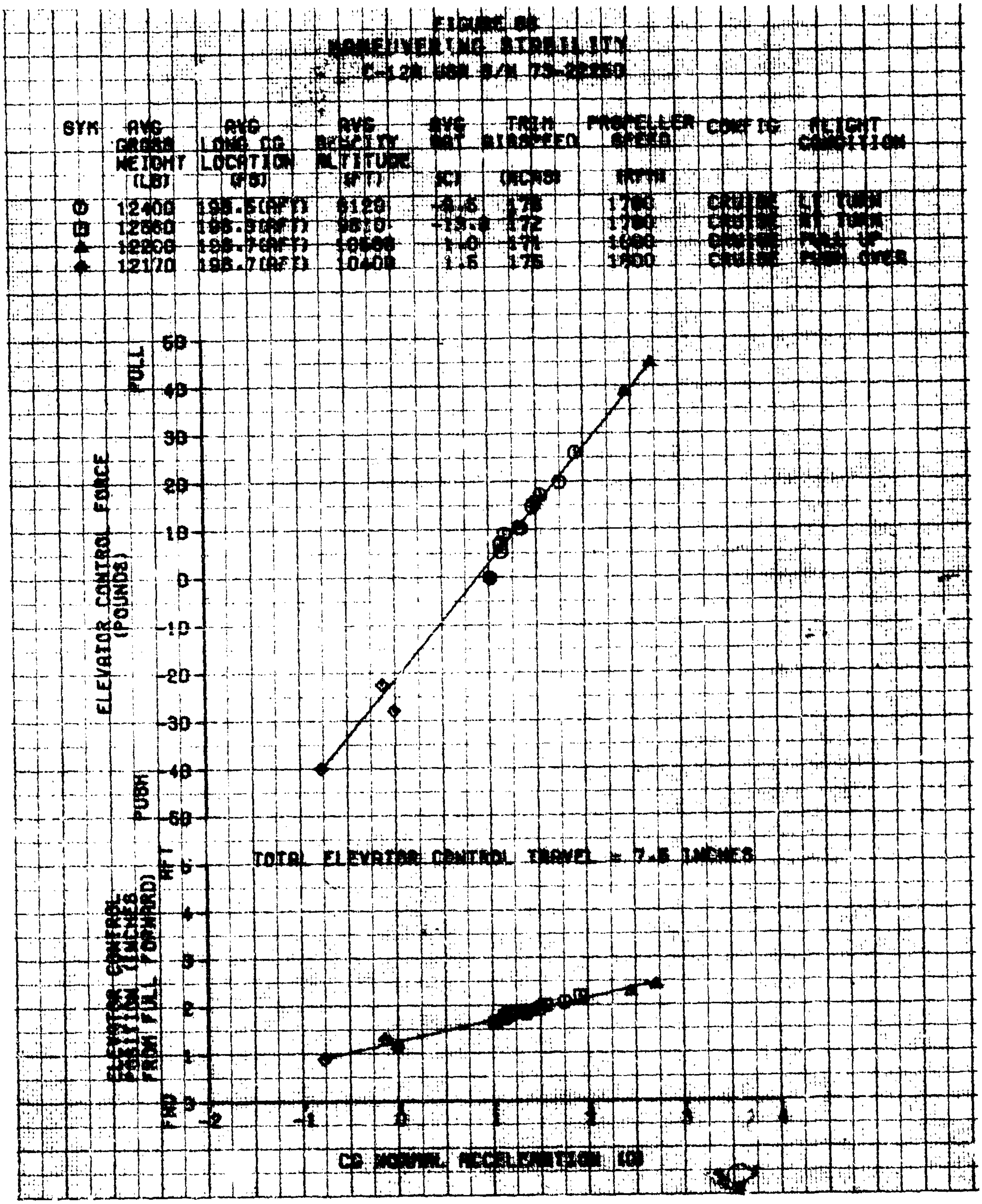


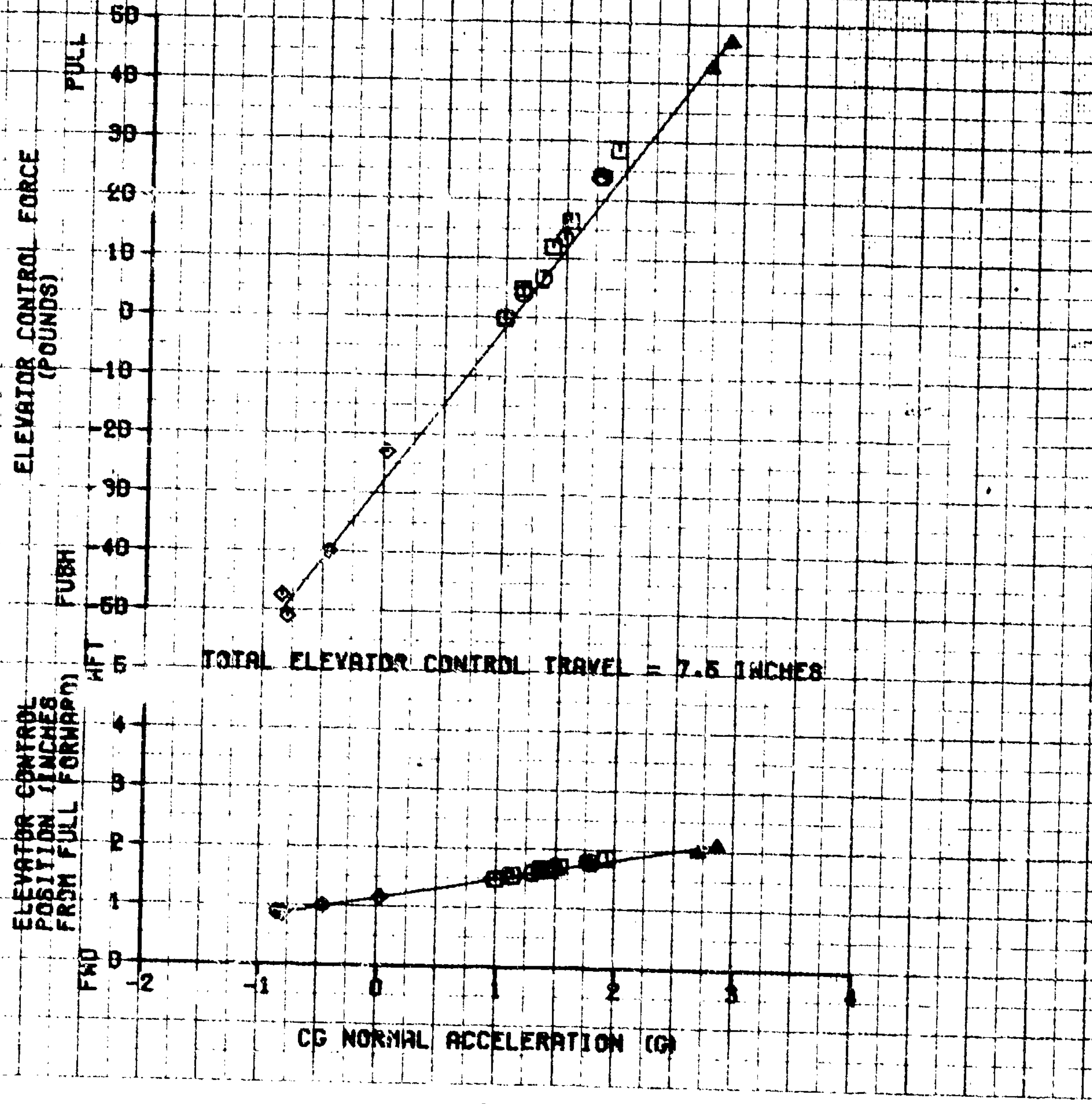
FIGURE 86
MANEUVERING STABILITY
C-12A USA S/N 73-22250
SYM
AVG GROSS WEIGHT (LB)
AVG LONG CG LOCATION (FS)
AVG DENSITY ALTITUDE (FT)
AVG OAT (C)
TRIM AIRSPEED (KCAS)
PROPELLER SPEED (RPM)
CONFIG
FLIGHT CONDITION
11820 197.0(AFT) 9150 -12.0 147 1790 CRUISE LT TURN
9470 -12.5 1790 CRUISE RT TURN
142 CRUISE PULL UP
11068 145 1800 CRUISE PUSH OVER
ELEVATOR CONTROL FORCE (POUNDS)
PULL
PUSH
50
40
30
20
10
0
-10
-20
-30
-40
-50
TOTAL ELEVATOR CONTROL TRAVEL = 7.5 INCHES
ELEVATOR CONTROL POSITION (INCHES FROM FULL FORWARD)
AFT
FWD
5
4
3
2
1
0
-2
-1
0
1
2
3
4
CG NORMAL ACCELERATION (G)



MANEUVERING STABILITY
SYM
AVG
GROSS
WEIGHT
(LB)
AVG
LONG CG
LOCATION
AVG
ALTITUDE
(FT)
AVG
TRIM
AIRSPEED
PROPELLER
SPEED
CONFIG
FLIGHT
CONDITION
12400
12660
12200
12170
1.5
175
1700
ELEVATOR CONTROL FORCE
(POUNDS)
PULL
PUSH
TOTAL ELEVATOR CONTROL TRAVEL = 7.5 INCHES
AFT
FWD
CG NORMAL ACCELERATION (G)

FIGURE 87
MANEUVERING STABILITY
C-12A USA S/N 73-22250

SYM	AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	TRIM AIRSPEED (KCAS)	PROPELLER SPEED (RPM)	CONFIG	FLIGHT CONDITION
○	12420	195.4(AFT)	12120	+1.0	209	1800	CRUISE	LT TURN
□	12620	195.3(AFT)	11530	+0.5	208	1800	CRUISE	RT TURN
▲	12200	195.8(AFT)	12600	-1.5	208	1800	CRUISE	PULL UP
◆	12250	195.5(AFT)	11920	0.5	209	1800	CRUISE	PUSH OVER



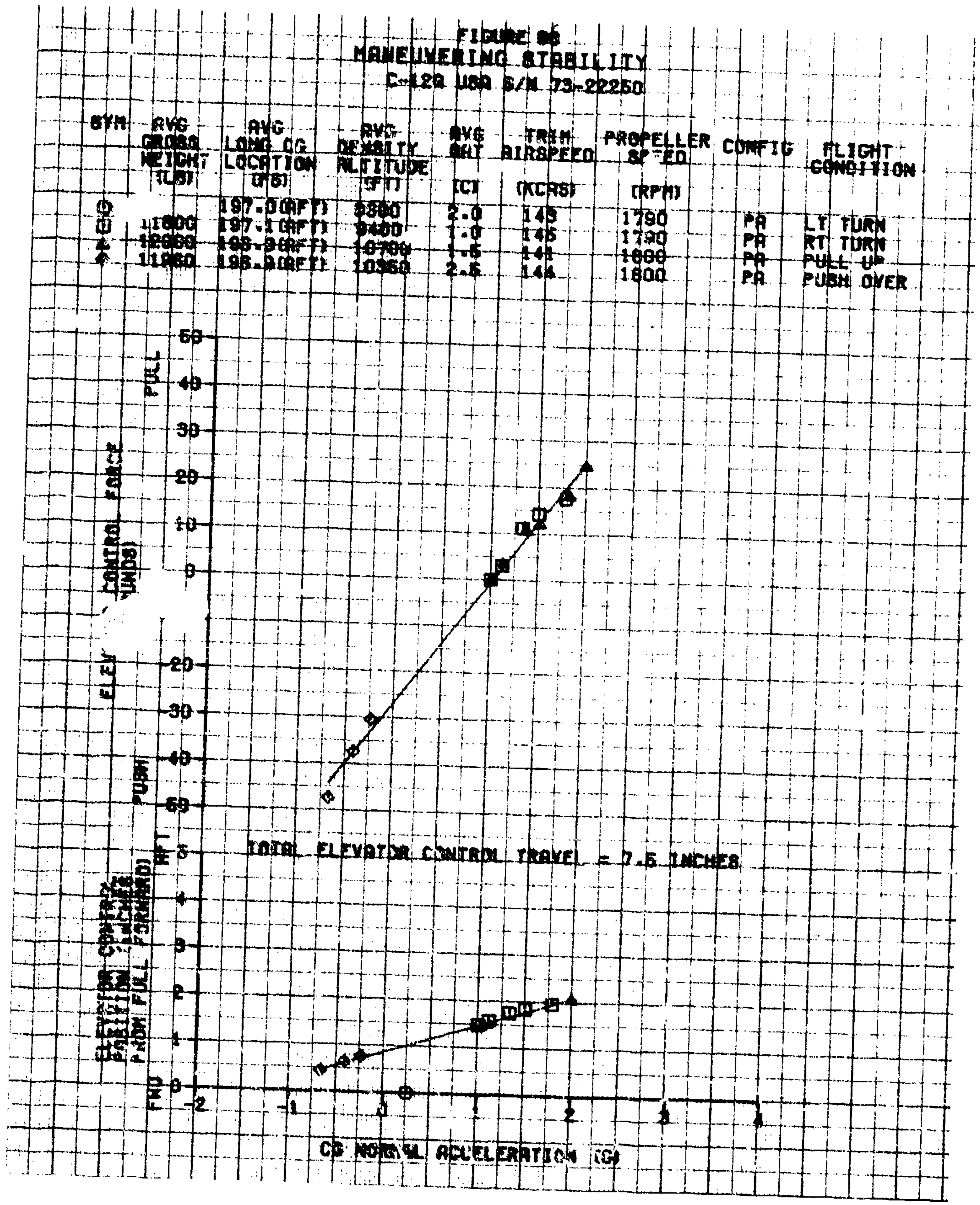


FIGURE 86
MANEUVERING STABILITY
C-12A USA S/N 73-22250
SYM
AVG GROSS WEIGHT (LB)
AVG LONG CG LOCATION (FS)
AVG DENSITY ALTITUDE (FT)
AVG OAT (C)
TRIM AIRSPEED (KCAS)
PROPELLER SPEED (RPM)
CONFIG
FLIGHT CONDITION
197.0(AFT) 9800 2.0 143
11800 197.1(AFT) 9400 1.0 145 1790 PA LT TURN
12000 198.9(AFT) 10700 1.5 141 1790 PA RT TURN
11960 198.9(AFT) 10360 2.5 144 1800 PA PULL UP
1800 PA PUSH OVER
PULL
PUSH
ELEVATOR CONTROL FORCE (POUNDS)
TOTAL ELEVATOR CONTROL TRAVEL = 7.5 INCHES
AFT
FWD
ELEVATOR CONTROL POSITION (INCHES FROM FULL FORWARD)
CG NORMAL ACCELERATION (G)

FIGURE 80
MANEUVERING STABILITY
C-12A USA S/N 73-22250

SYM	AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM AIRSPEED (KCAS)	PROPELLER SPEED (RPM)	CONFIG	FLIGHT CONDITION
○	12100.	[illegible](AFT)	[illegible]	+12.0	[illegible]	1700	[illegible]	LT TURN
□	12240.	196.7(AFT)	[illegible]	+12.6	[illegible]	1700	[illegible]	RT TURN
△	12140	[illegible](AFT)	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	PULL UP
◇	12080	[illegible](AFT)	[illegible]	1.5	[illegible]	[illegible]	[illegible]	PUSH OVER

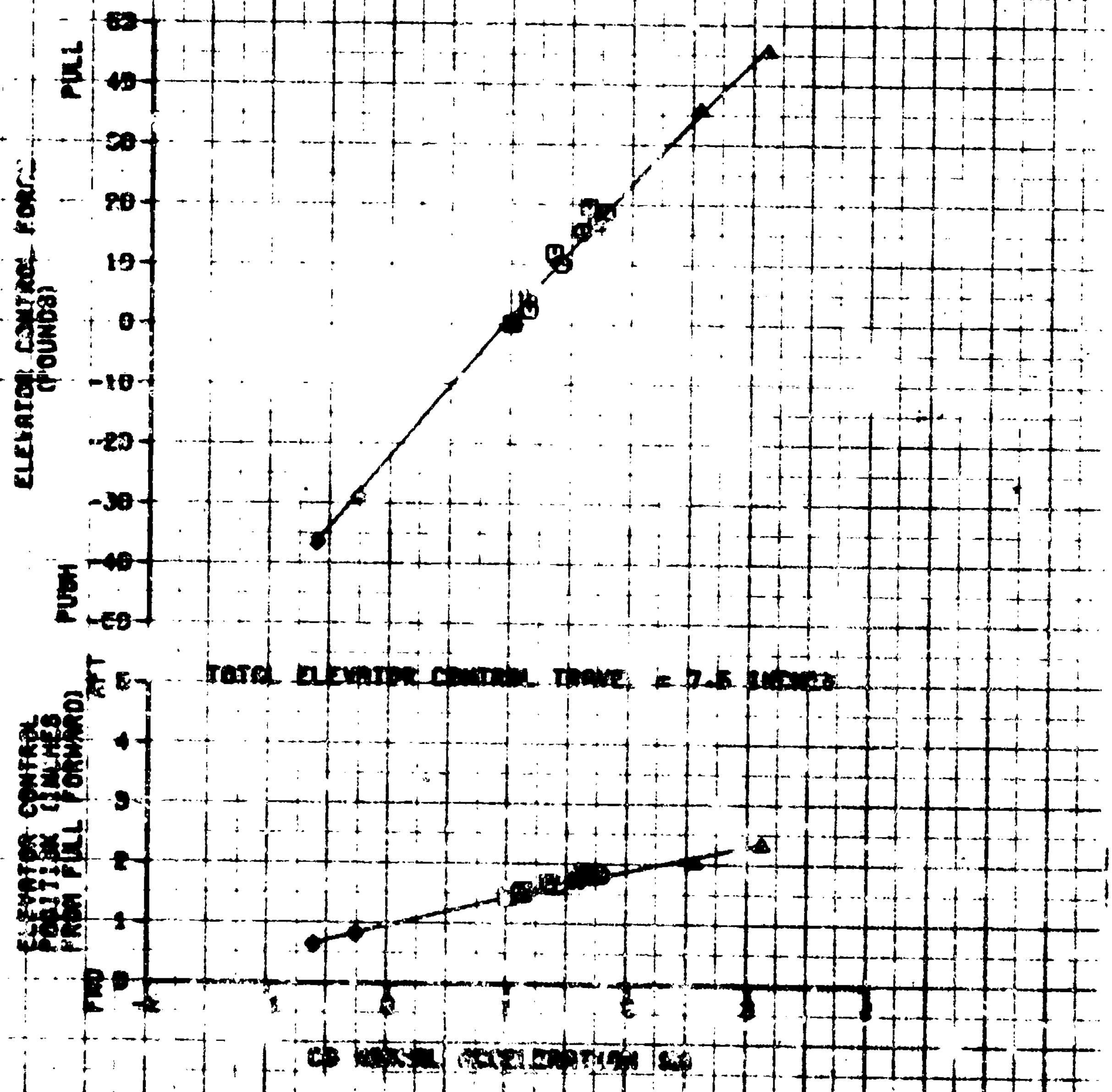


FIGURE 90
ROLL PERFORMANCE
C-12A USA S/N 73-22250
ENGINE MODEL PT6A-38

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	PROPELLER SPEED (RPM)	TRIM AIRSPEED (KCAS)	CONFIGURATION	FLIGHT CONDITION
12300	186.1(FWD)	10140	-7.5	1800	124	POWER APPROACH	LEVEL FLIGHT

SOLID LINE

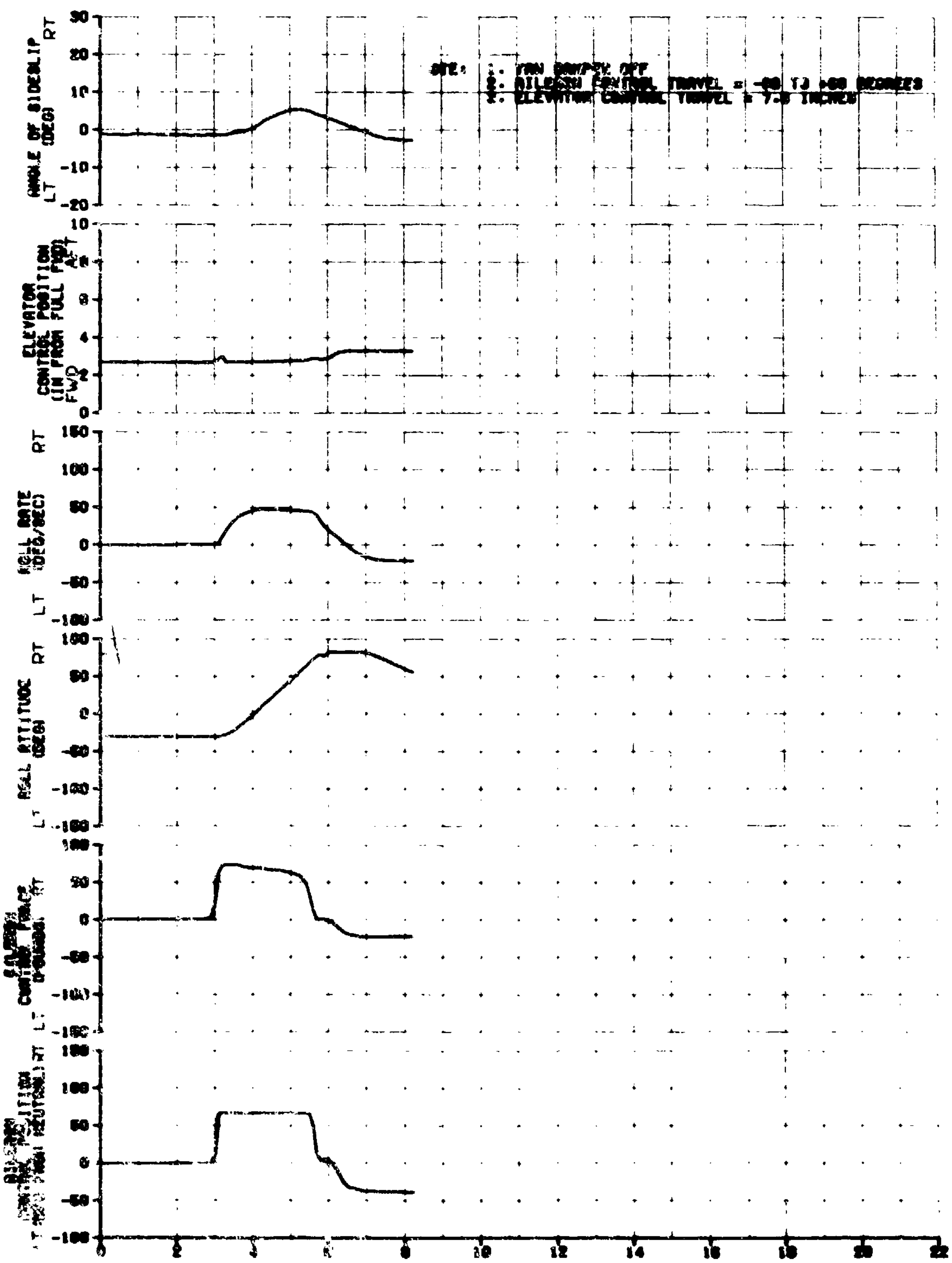


FIGURE 91
ROLL PERFORMANCE
C-12A USA S/N 73-22250
ENGINE MODEL PT6A-38

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	PROPELLER SPEED (RPM)	TRIM AIRSPEED (KCAS)	CONFIGURATION	FLIGHT CONDITION
12250	185.1(FWD)	10240	-8.0	1800	124	POWER APPROACH	LEVEL FLIGHT

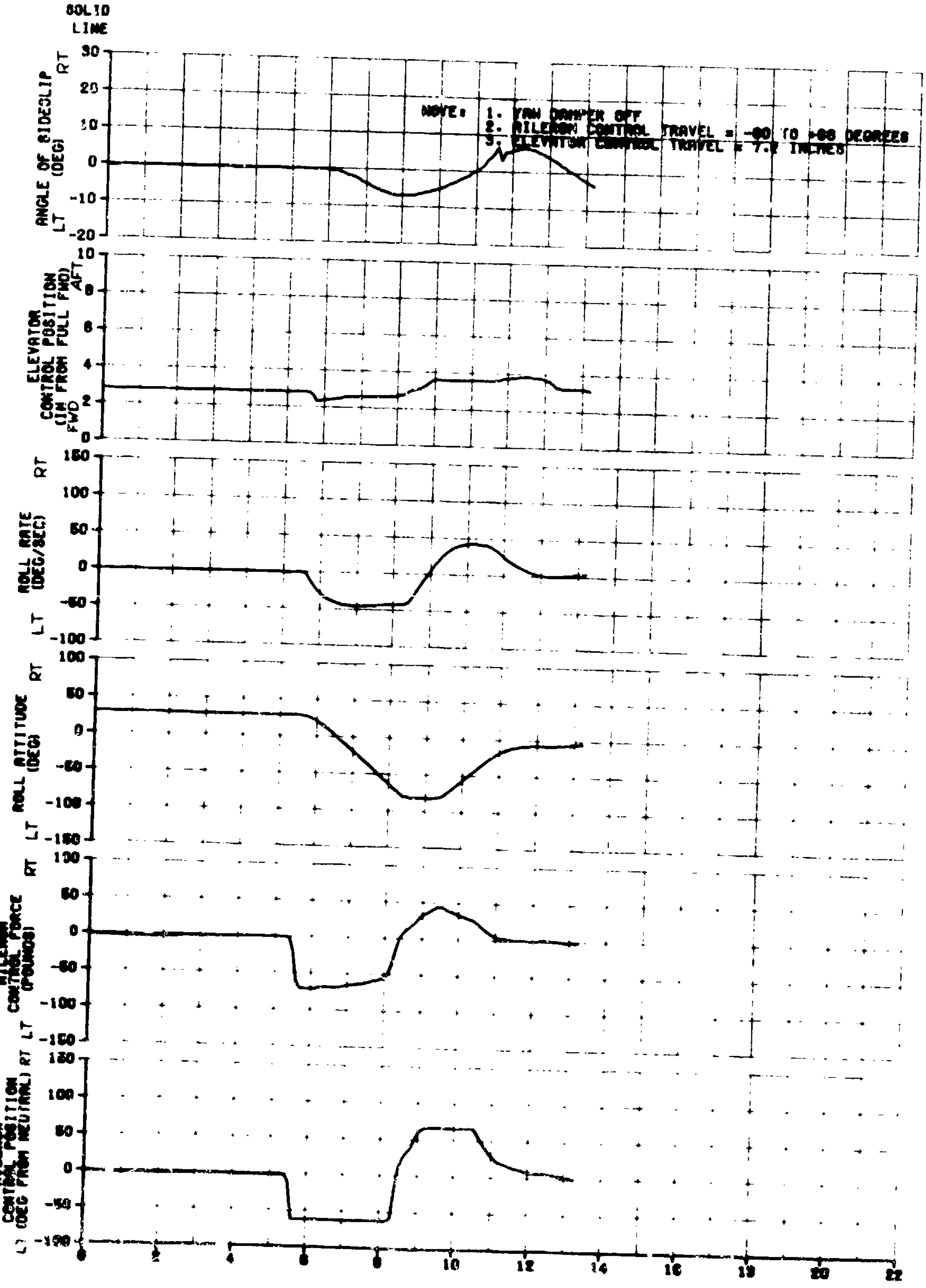


FIGURE 9?

ROLL PERFORMANCE

C-12A USA S/N 73-22250

ENGINE MODEL PT6A-38

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	PROPELLER SPEED (RPM)	TRIM AIRSPEED (KCAS)	CONFIGURATION	FLIGHT CONDITION
11920	185.8(MID)	10340	-8.5	1800	185	POWER APPROACH	LEVEL FLIGHT

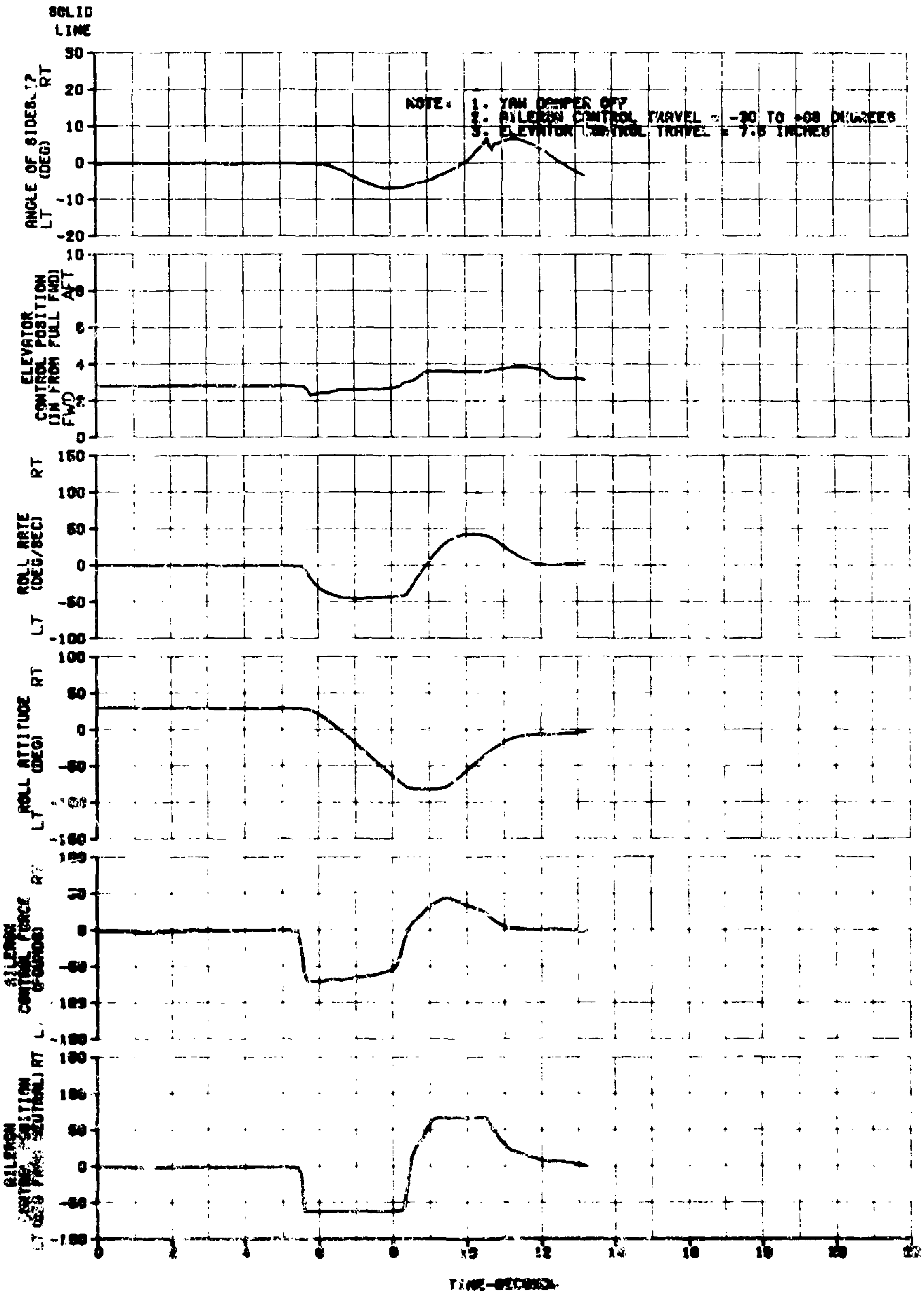


FIGURE 93
ROLL PERFORMANCE
C-12A USA S/N 73-22250
ENGINE MODEL PT6A-38

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	PROPELLER SPEED (RPM)	TRIM AIRSPEED (KCAS)	CONFIGURATION	FLIGHT CONDITION
12170	185.1(FWD)	9970	-8.0	1900	124	POWER APPROACH	LEVEL FLIGHT

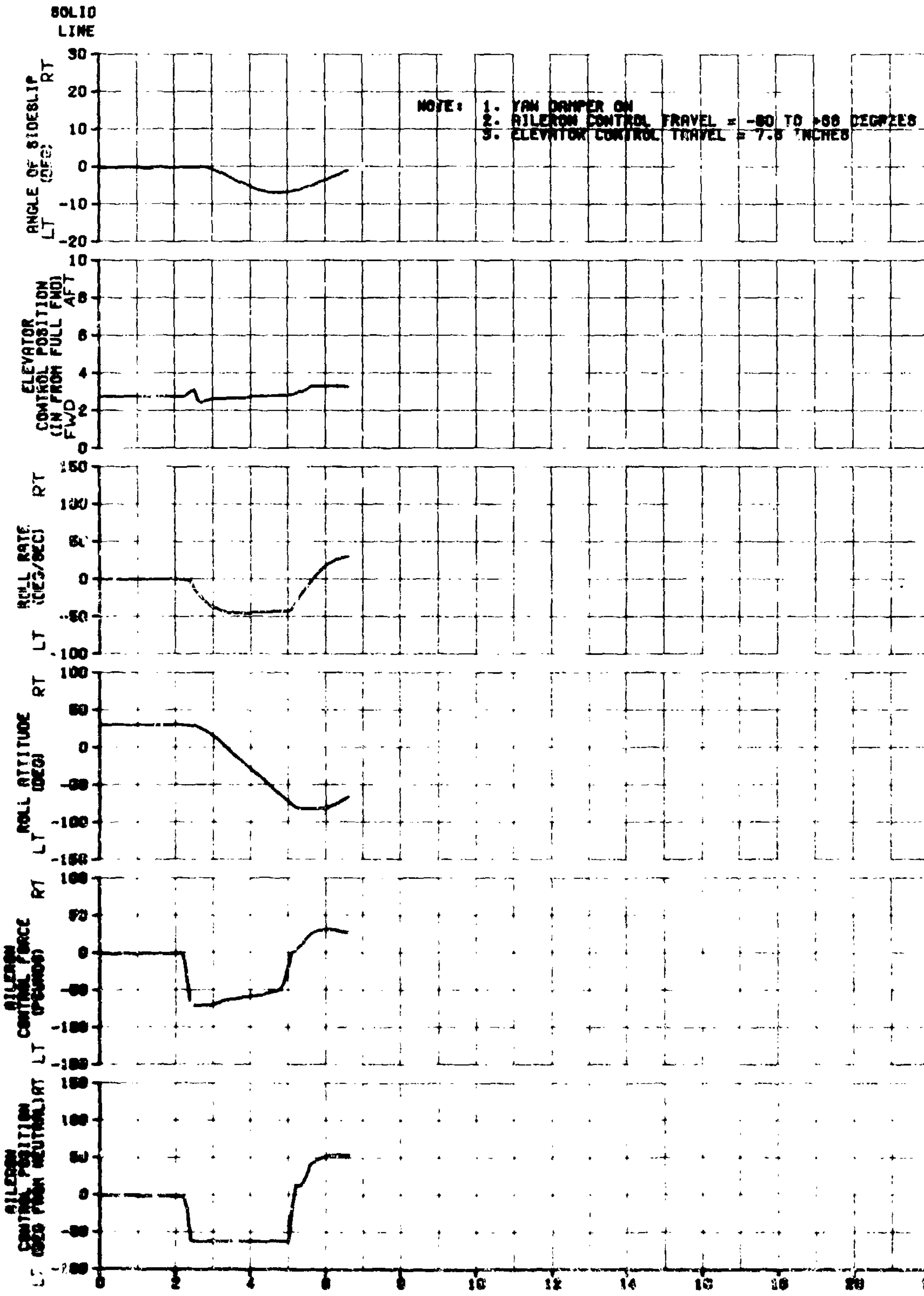


FIGURE 94
ROLL PERFORMANCE
C-12A USA S/N 73-22250
ENGINE MODEL PT6A-38

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	PROPELLER SPEED (RPM)	TRIM AIRSPEED (KCAS)	CONFIGURATION	FLIGHT CONDITION
12030	186.1(FWD)	10510	-8.5	1900	157	POWER APPROACH	LEVEL FLIGHT

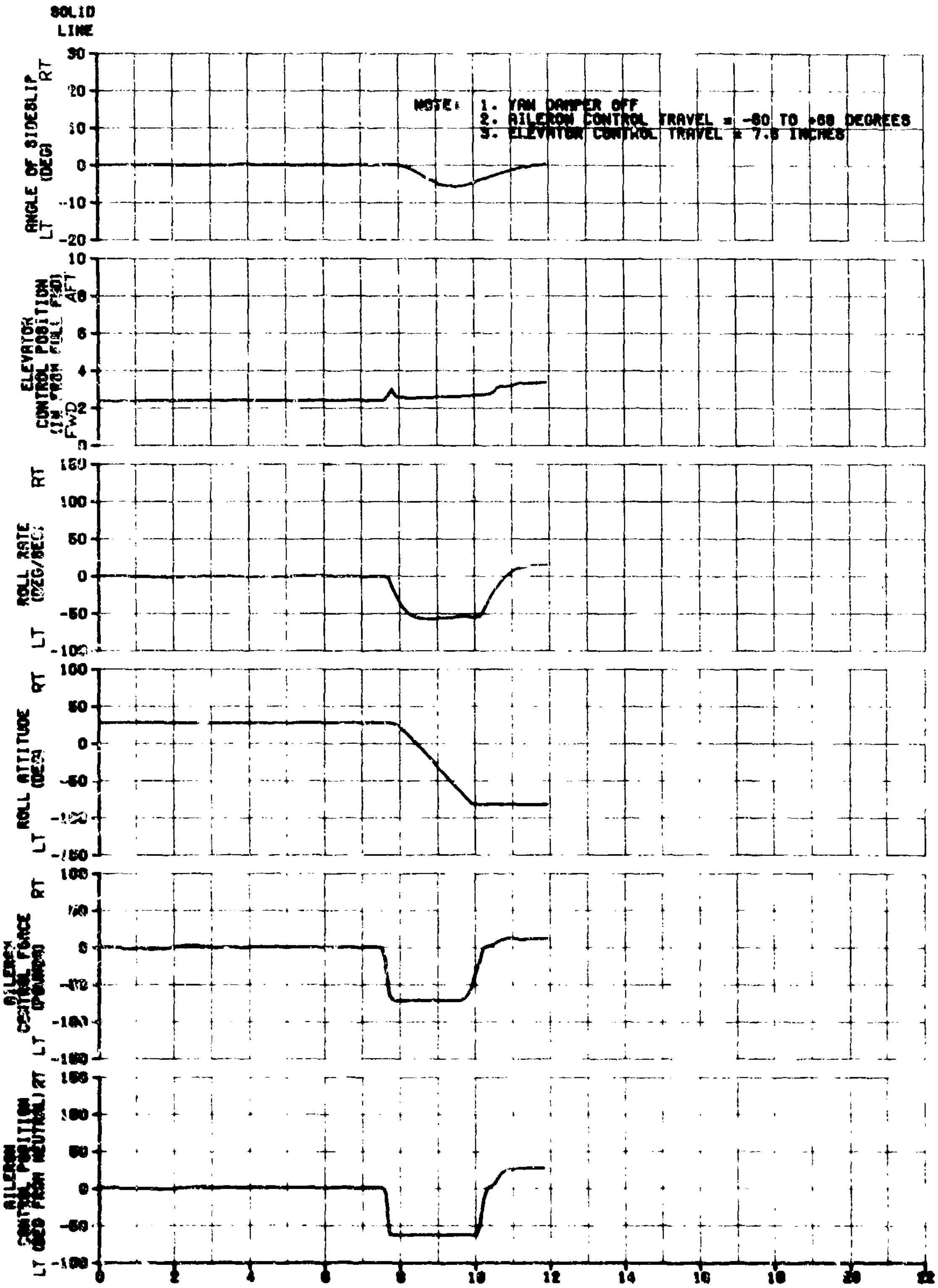


FIGURE 95
DYNAMIC VMC
C-12A USA S/N 73-22250

GROSS WEIGHT (LB)	LONG CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (C)	PROPELLER SPEED (RPM)	TRIM AIRSPEED (KCAS)	CONFIGURATION	FLIGHT CONDITION
11060	196.2 (AFT)	11610	-2.0	2000	93	CRUISE	LEVEL FLIGHT

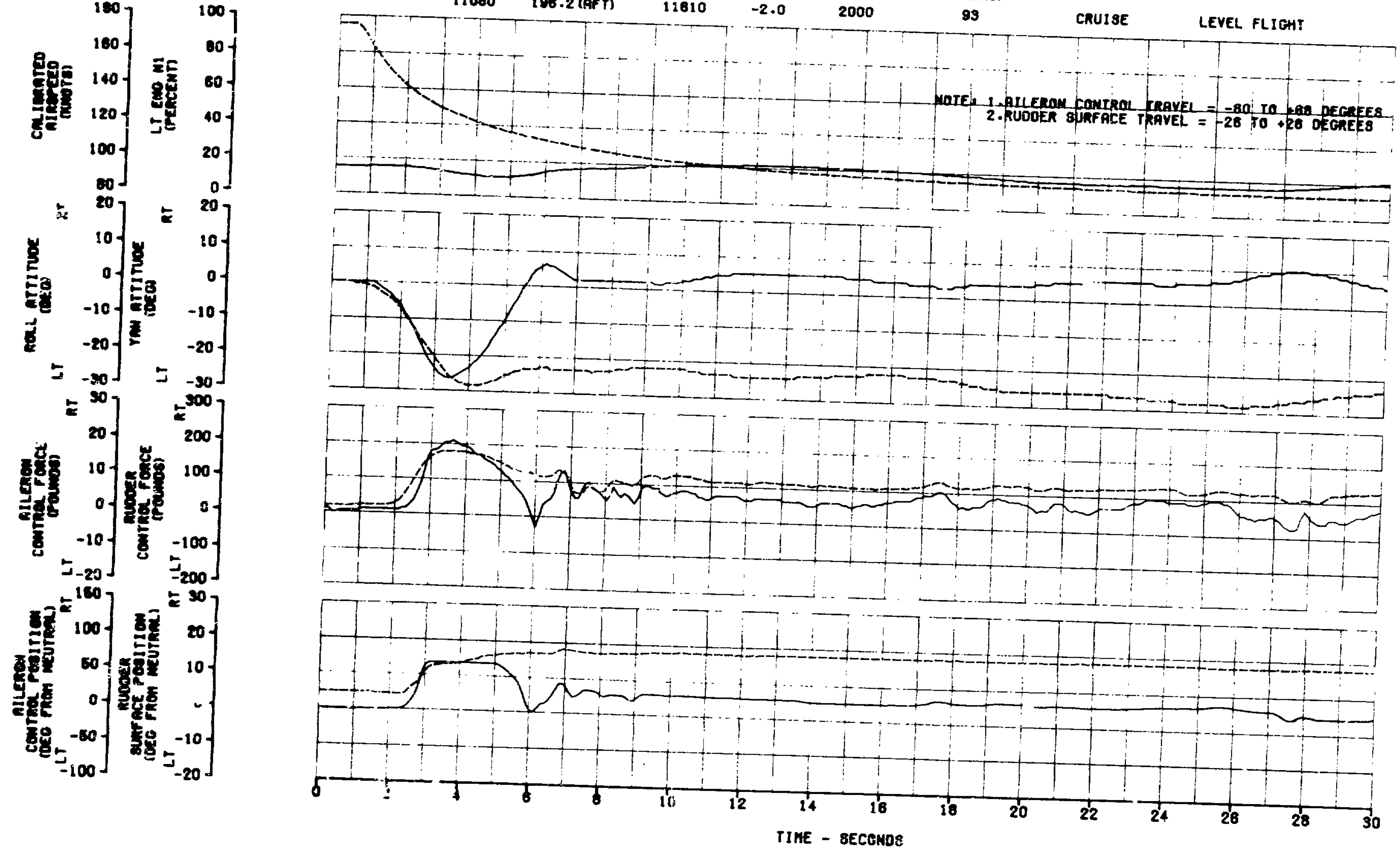


FIGURE 98
DYNAMIC VMC
C-12A USA S/N 73-22250

GROSS WEIGHT (LB)	LONG CG LOCATION (FS)	DENSITY ALTITUDE (FT)	OAT (C)	PROPELLER SPEED (RPM)	TRIM AIRSPEED (KCAS)	CONFIGURATION	FLIGHT CONDITION
12130	198.8 (AFT)	7700	1.5	2000	92	WAVEOFF	CLIMB

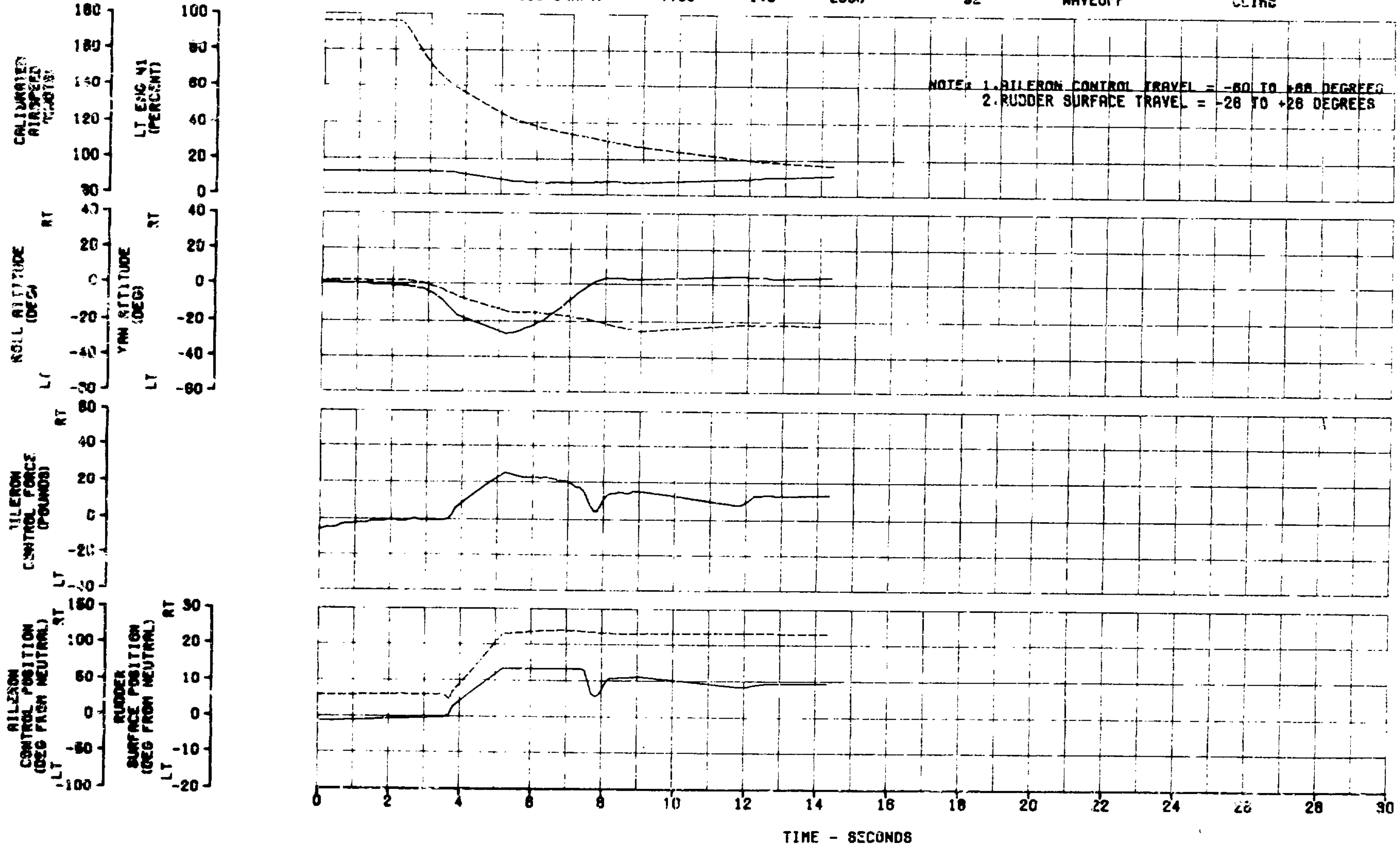
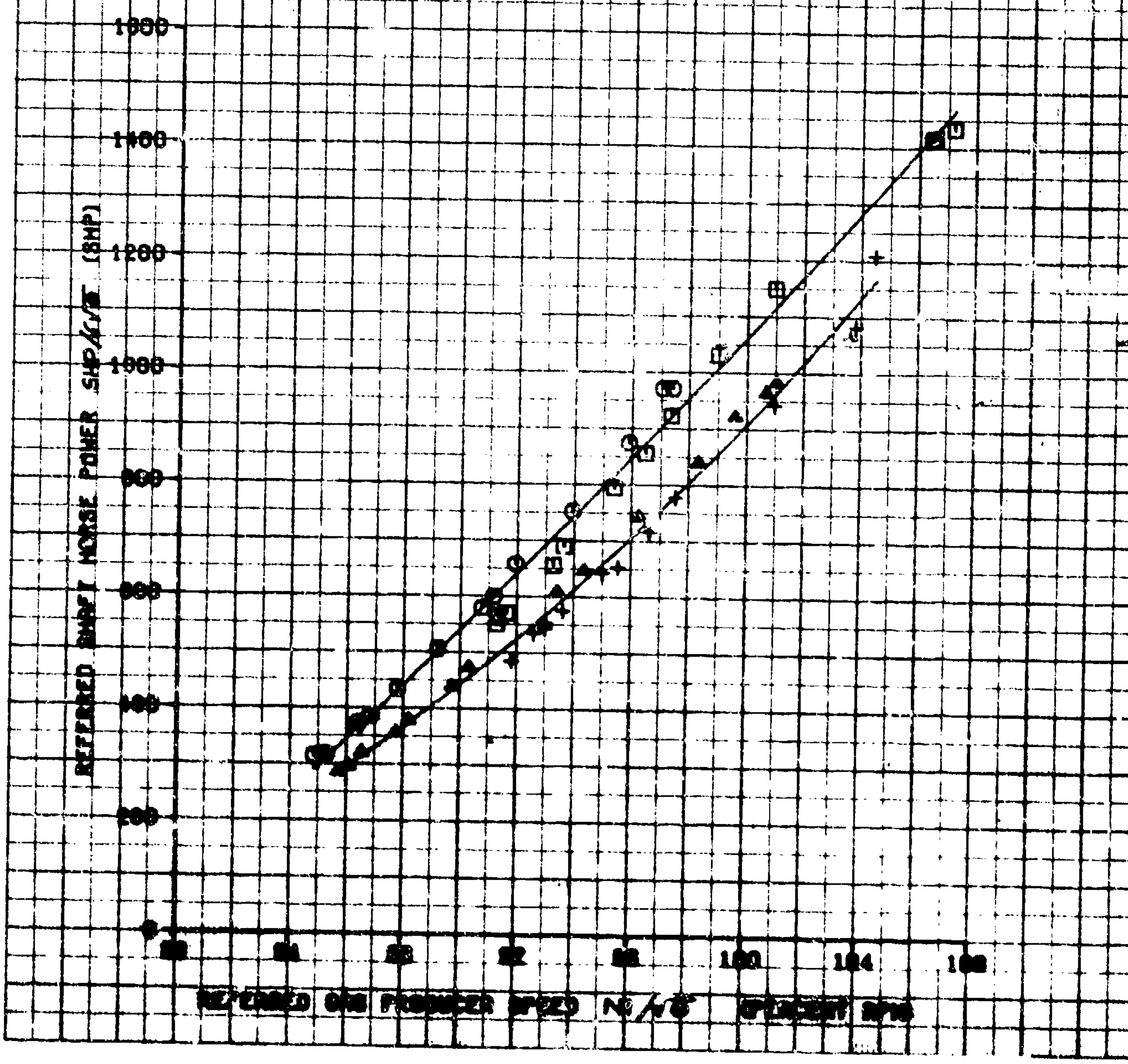


FIGURE 97
ENGINE CHARACTERISTICS
C-12A USA S/N 73-22250
ENGINE MODEL PT6A-38 S/N PC-E-79C03

SYM	AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	PROPELLER SPEED (RPM)	CONFIGURATION	FLIGHT CONDITION
⊙	12460.	184.6(FWD)	11290.	6.5	1800.	CRUISE	LEVEL FLIGHT
⊡	12400.	184.5(FWD)	20050.	-17.6	1800.	CRUISE	LEVEL FLIGHT
▲	11020.	186.2(FWD)	10050.	-4.9	1790.	CRUISE (ICE VANES)	LEVEL FLIGHT
+	12120.	184.8(FWD)	20030.	-25.1	1780.	CRUISE (ICE VANES)	LEVEL FLIGHT



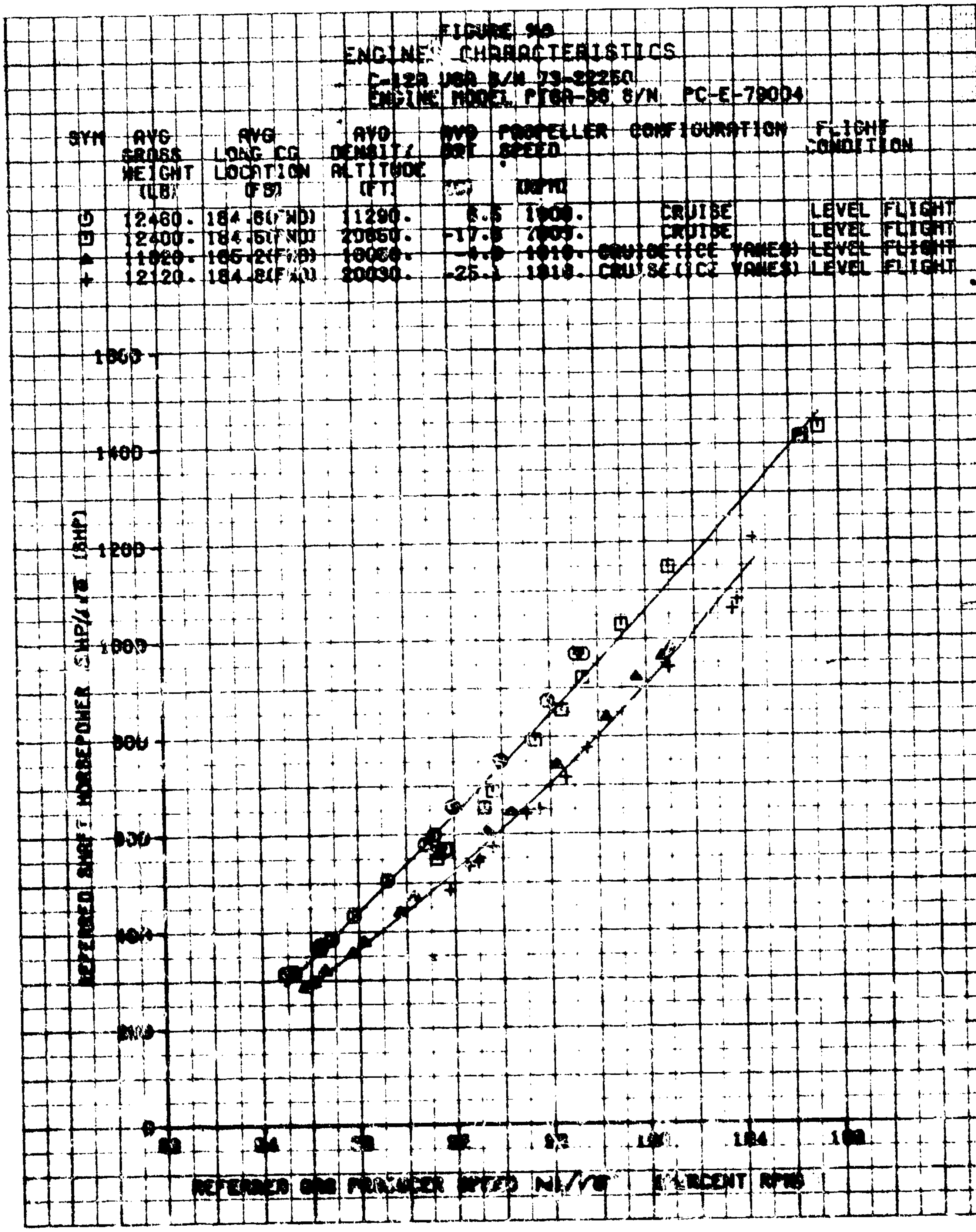

ENGINE CHARACTERISTICS
SYM
AVG GROSS WEIGHT (LB)
AVG LONG CG LOCATION (FS)
DENSITY ALTITUDE (FT)
PROPELLER SPEED (RPM)
CONFIGURATION
FLIGHT CONDITION
12460.
12400.
12120.
11290.
20030.
CRUISE
CRUISE
LEVEL FLIGHT
LEVEL FLIGHT
LEVEL FLIGHT
LEVEL FLIGHT
1400
1200
1000
0
REFERRED SHAFT HORSEPOWER
PERCENT RPM

FIGURE 99
ENGINE CHARACTERISTICS

C-12A USA S/N 73-22250
ENGINE MODEL PT6A-38 S/N PC-E-79003

SYM	AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	PROPELLER SPEED (RPM)	CONFIGURATION	FLIGHT CONDITION
○	12460.	184.6(FWD)	11290.	6.5	1800.	CRUISE	LEVEL FLIGHT
□	12400.	184.5(FWD)	20850.	-17.8	1800.	CRUISE	LEVEL FLIGHT
△	11020.	[illegible]	[illegible]	-4.9	1790.	CRUISE(ICE VANES)	LEVEL FLIGHT
+	12120.	184.6(FWD)	20030.	-25.1	1780.	CRUISE(ICE VANES)	LEVEL FLIGHT

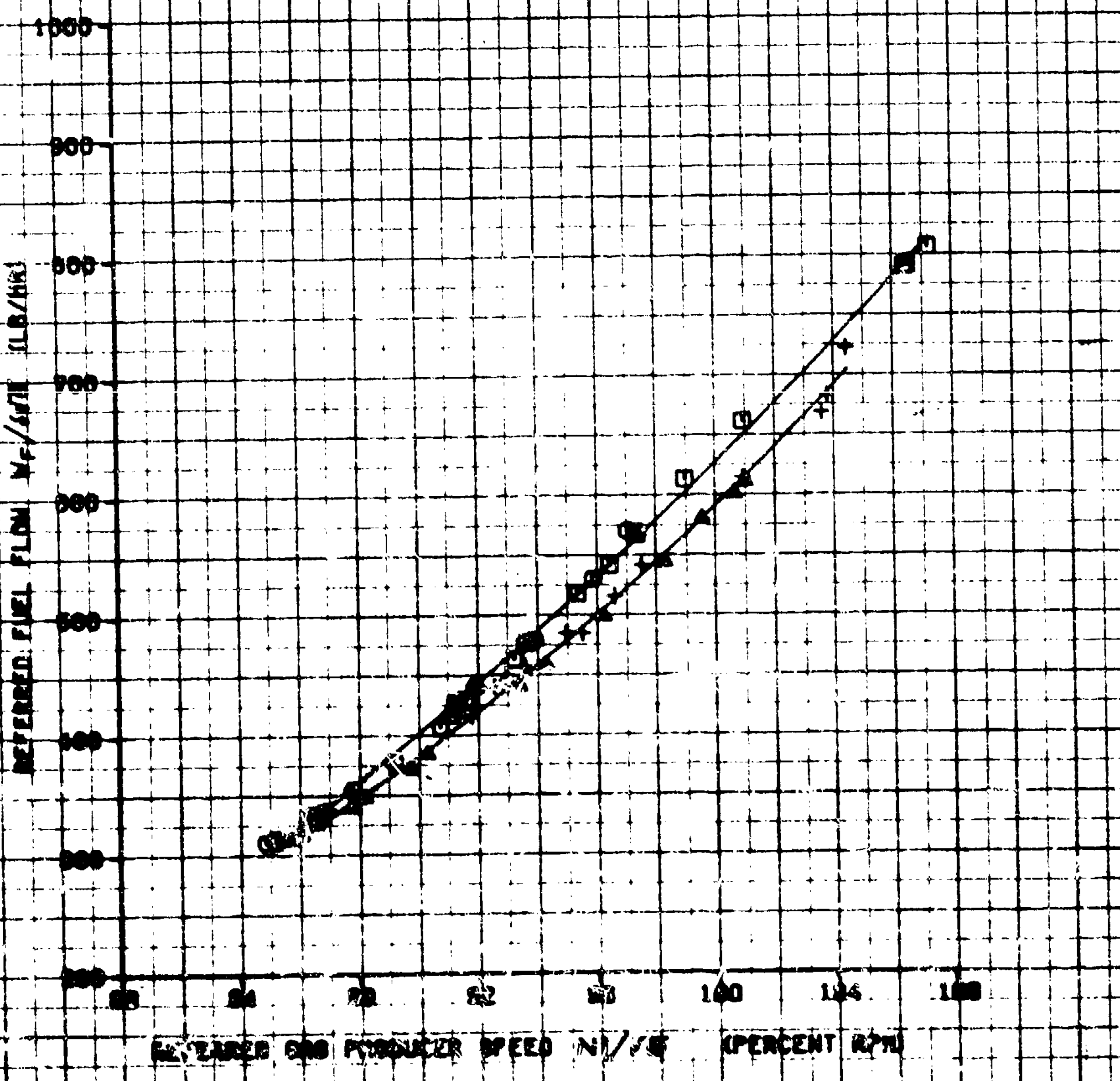
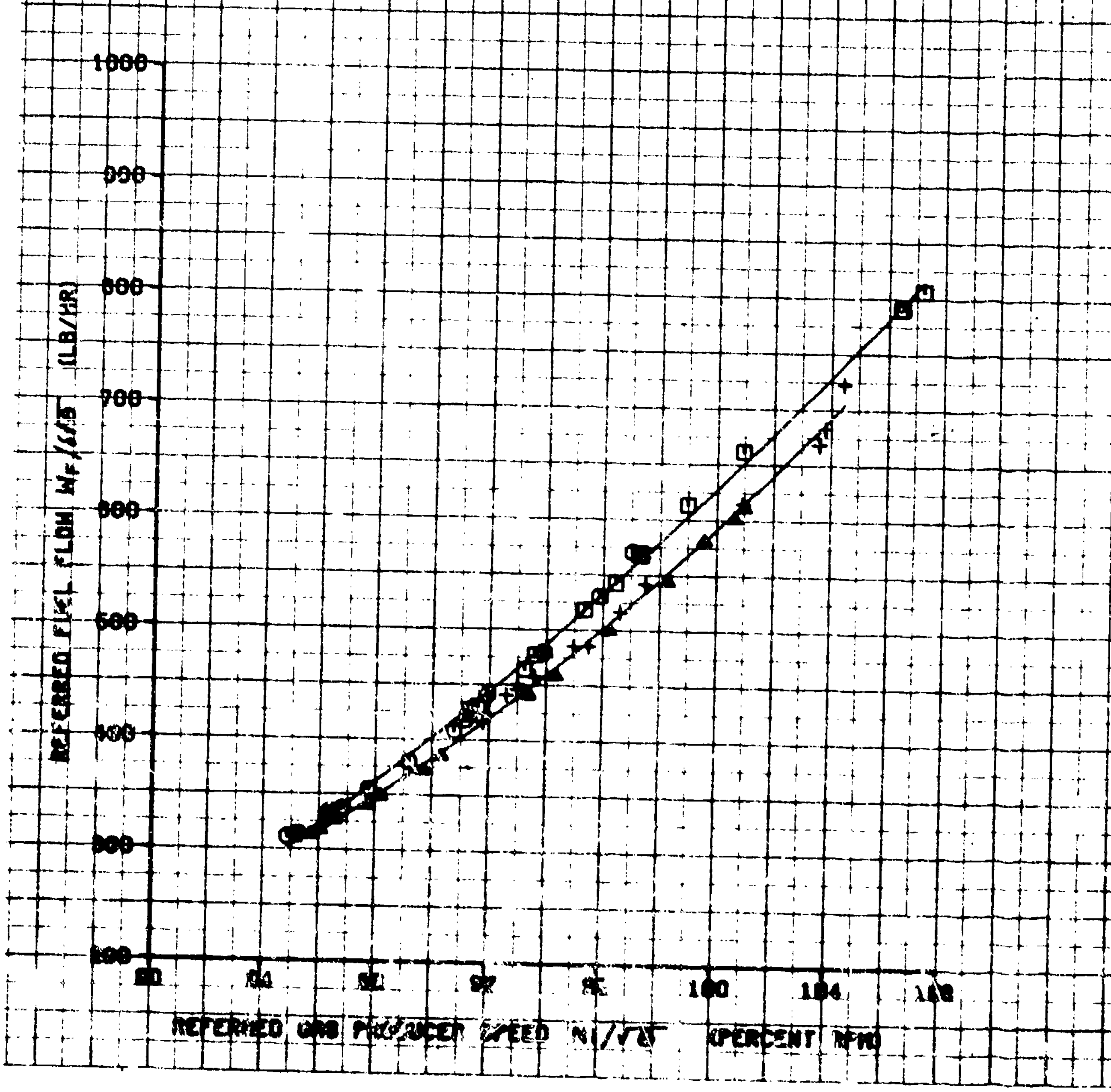


FIGURE 100
ENGINE CHARACTERISTICS
C-12A USA S/N 73-22250
ENGINE MODEL PT6A-38 S/N PC-E-79004

SYM	AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	PROPELLER SPEED (RPM)	CONFIGURATION	FLIGHT CONDITION
○	12460.	184.6(FWD)	11260.	6.5	1900.	CRUISE	LEVEL FLIGHT
□	12400.	184.5(FWD)	20860.	-17.5	1900.	CRUISE	LEVEL FLIGHT
△	11920.	185.2(FWD)	10050.	-4.0	1910.	CRUISE (ICE VANES)	LEVEL FLIGHT
+	12120.	184.8(FWD)	20030.	-25.1	1910.	CRUISE (ICE VANES)	LEVEL FLIGHT



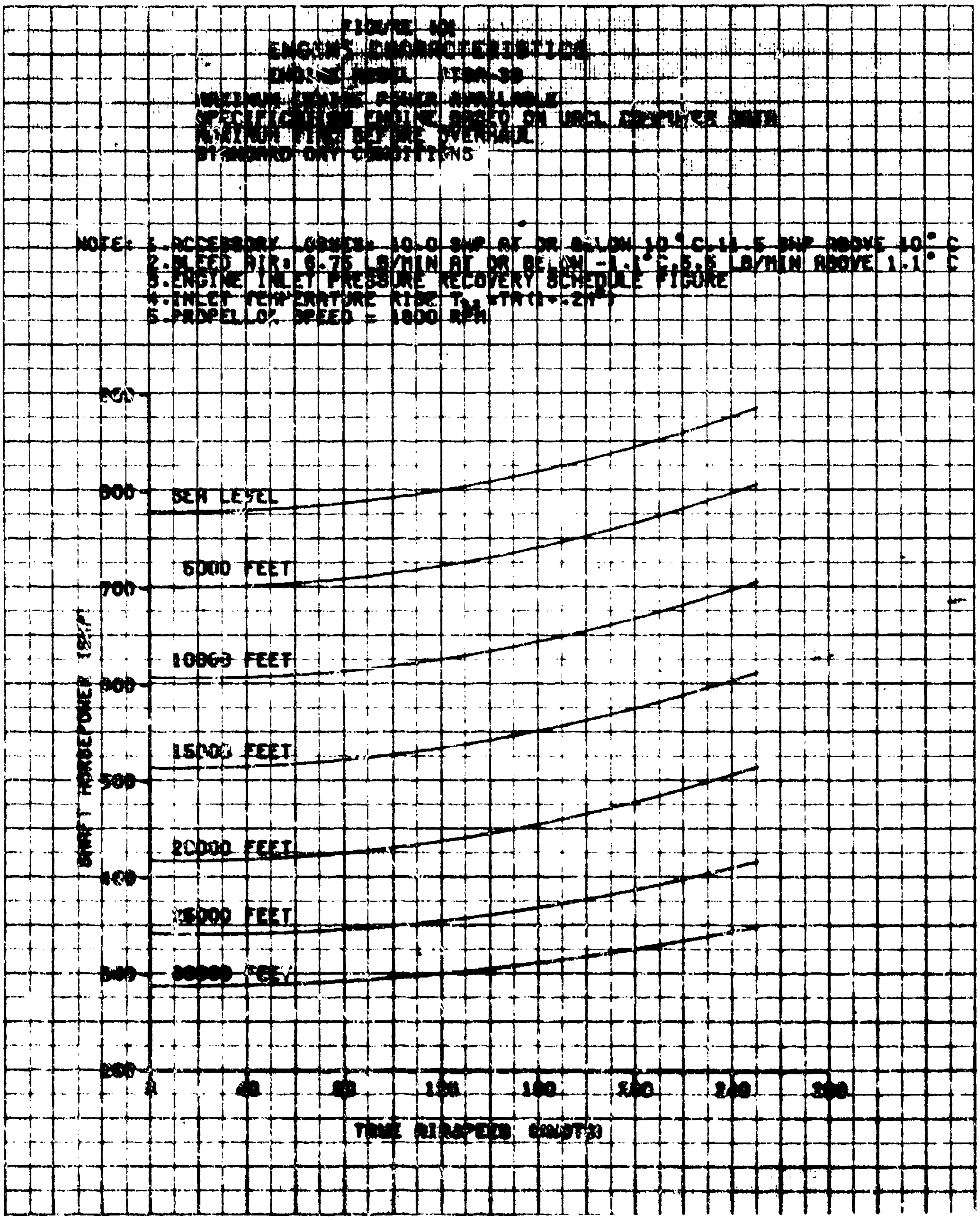

STANDARD DAY CONDITIONS
NOTE:
2.BLEED AIR: 6.75 LB/MIN AT OR BELOW
3.ENGINE INLET PRESSURE RECOVERY SCHEDULE FIGURE
4.INLET TEMPERATURE RISE
5.PROPELLOR SPEED = 1800 RPM
SEA LEVEL
5000 FEET
10000 FEET
15000 FEET
20000 FEET
25000 FEET
800
700
600
500
400
300
200
0
40
80
120
160
240
280
SHAFT HORSEPOWER
TRUE AIRSPEED (KNOTS)

FIGURE 102

ENGINE CHARACTERISTICS

ENGINE MODEL PT6A-38

SPECIFICATION ENGINE BASED ON UACL COMPUTER DATA
SPECIFICATION ENGINE IS A MINIMUM ENGINE WITH
MAXIMUM TIME BEFORE OVERHAUL
STANDARD DAY CONDITIONS

NOTE: 1. ACCESSORY LOSSES: 10.0 SHP AT OR BELOW 10° C, 11.5 SHP ABOVE 10° C
2. BLEED AIR: 6.75 LB/MIN AT OR BELOW -1.1° C, 8.5 LB/MIN ABOVE 1.1° C
3. ENGINE INLET PRESSURE RECOVERY SCHEDULE FIGURE
4. INLET TEMPERATURE RISE $T_{t2} = TA(1+.2M^2)$
5. PROPELLOR SPEED = 1800 RPM

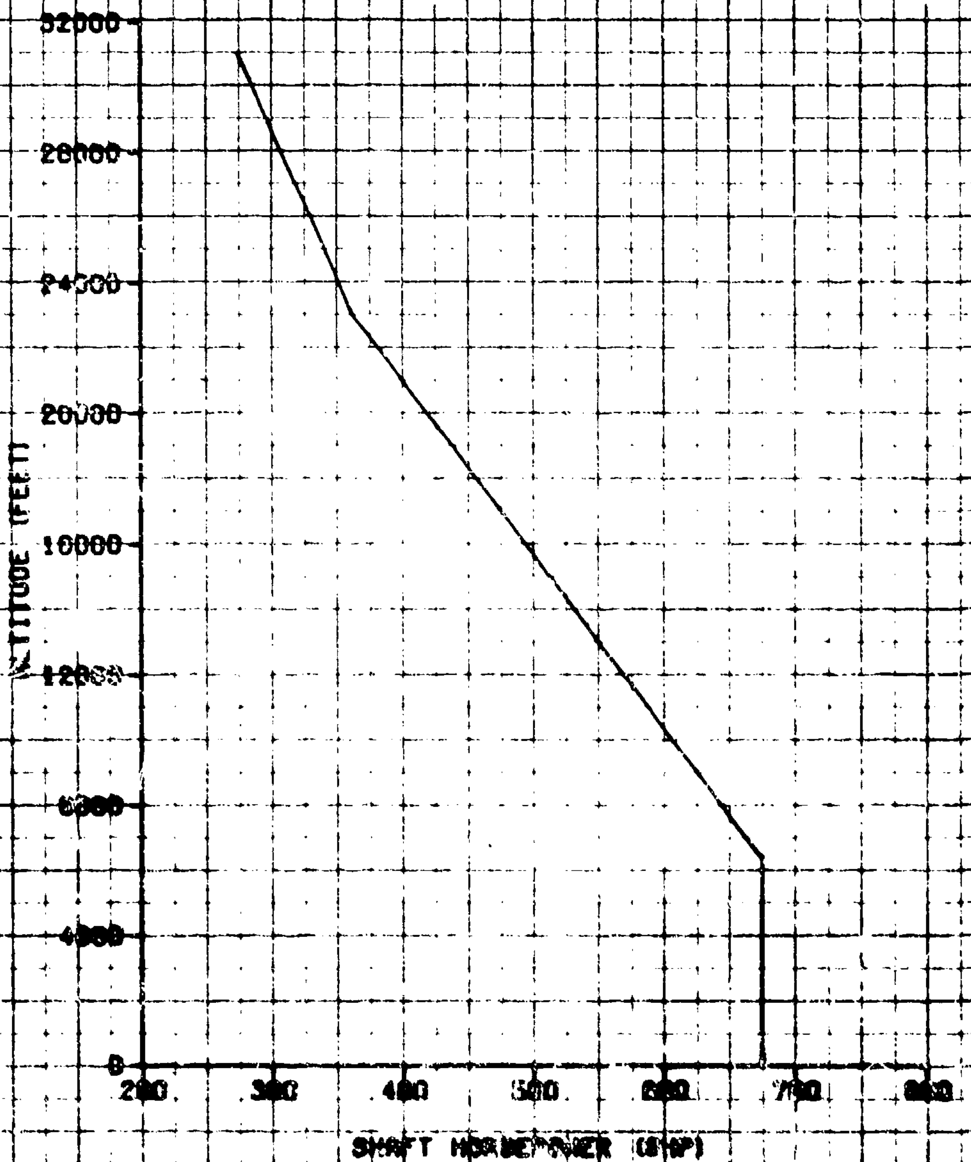


FIGURE 103
ENGINE CHARACTERISTICS
ENGINE MODEL PT6A-38
JET THRUST FOR MAXIMUM CRUISE POWER AVAILABLE
SPECIFICATION ENGINE BASED ON UACL COMPUTER DATA
MAXIMUM TIME BEFORE OVERHAUL
STANDARD DAY CONDITIONS

NOTE: 1. ACCESSORY LOSSES: 10.0 SHP AT OR BELOW 10° C, 11.5 SHP ABOVE 10° C
2. BLEED AIR: 6.75 LB/MIN AT OR BELOW -1.1° C, 5.6 LB/MIN ABOVE 1.1° C
3. ENGINE INLET PRESSURE RECOVERY SCHEDULE FIGURE
4. INLET TEMPERATURE RISE $T_{t2} = T_A(1+.2M^2)$
5. PROPELLOR SPEED - 1800 RPM

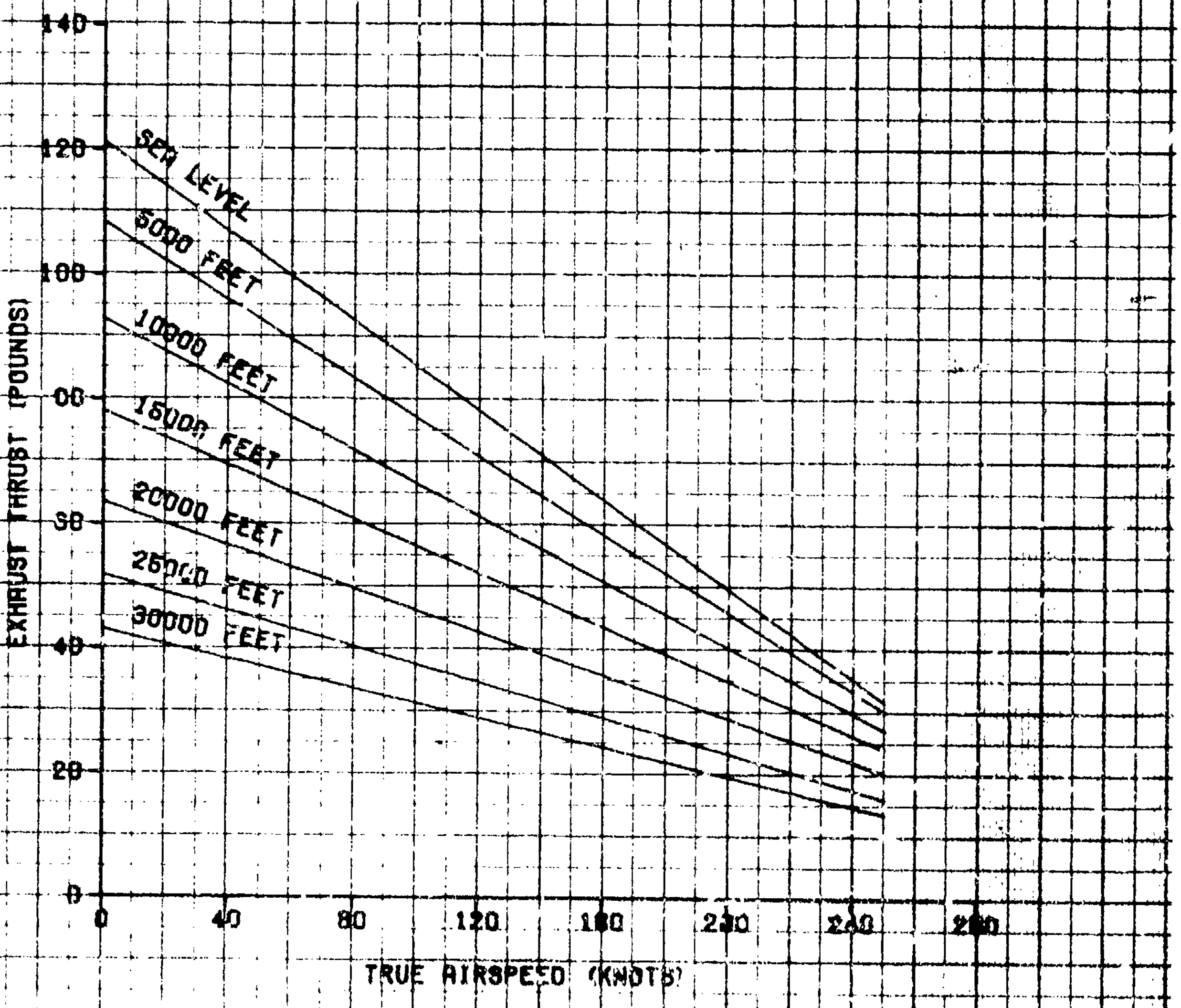
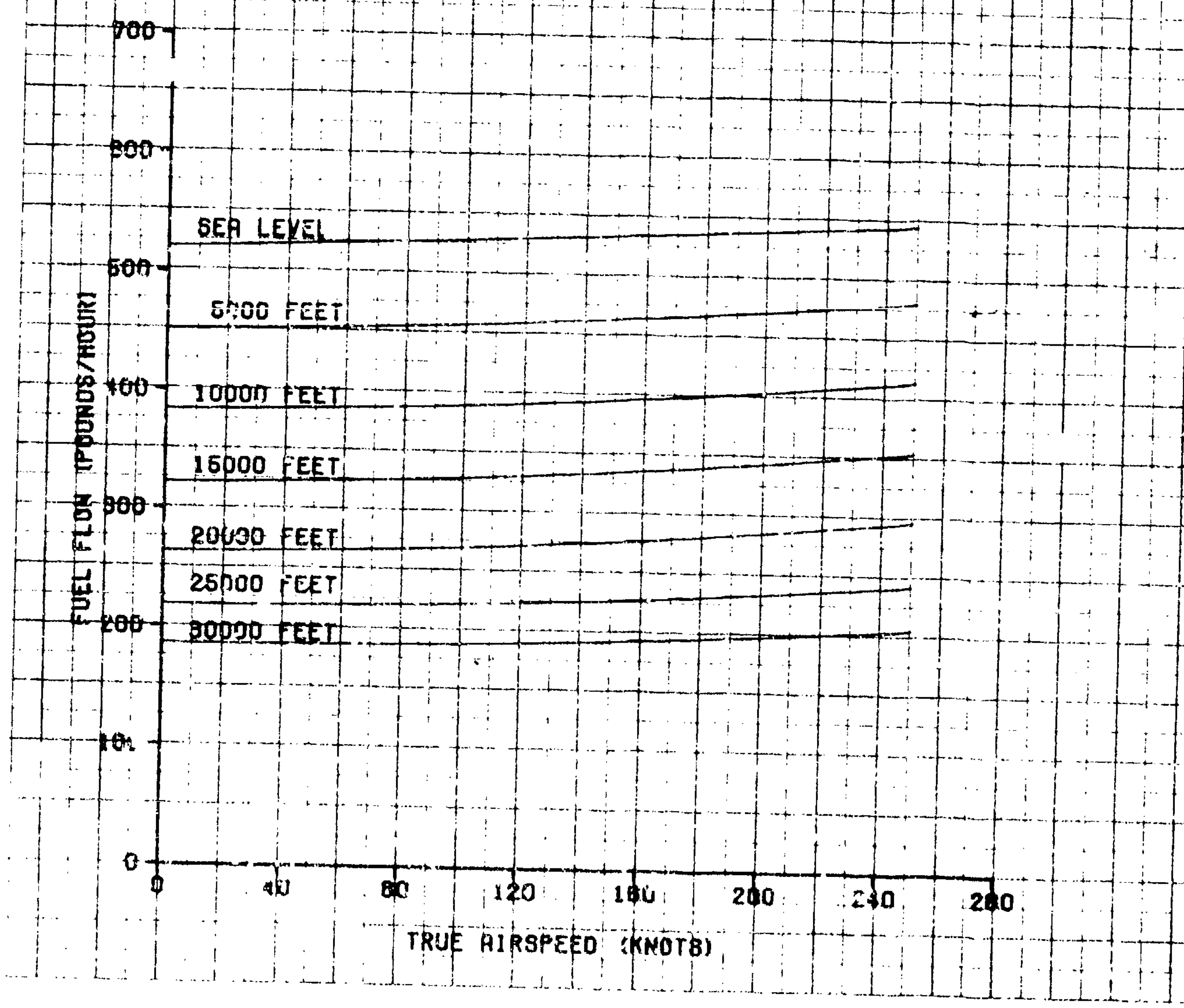
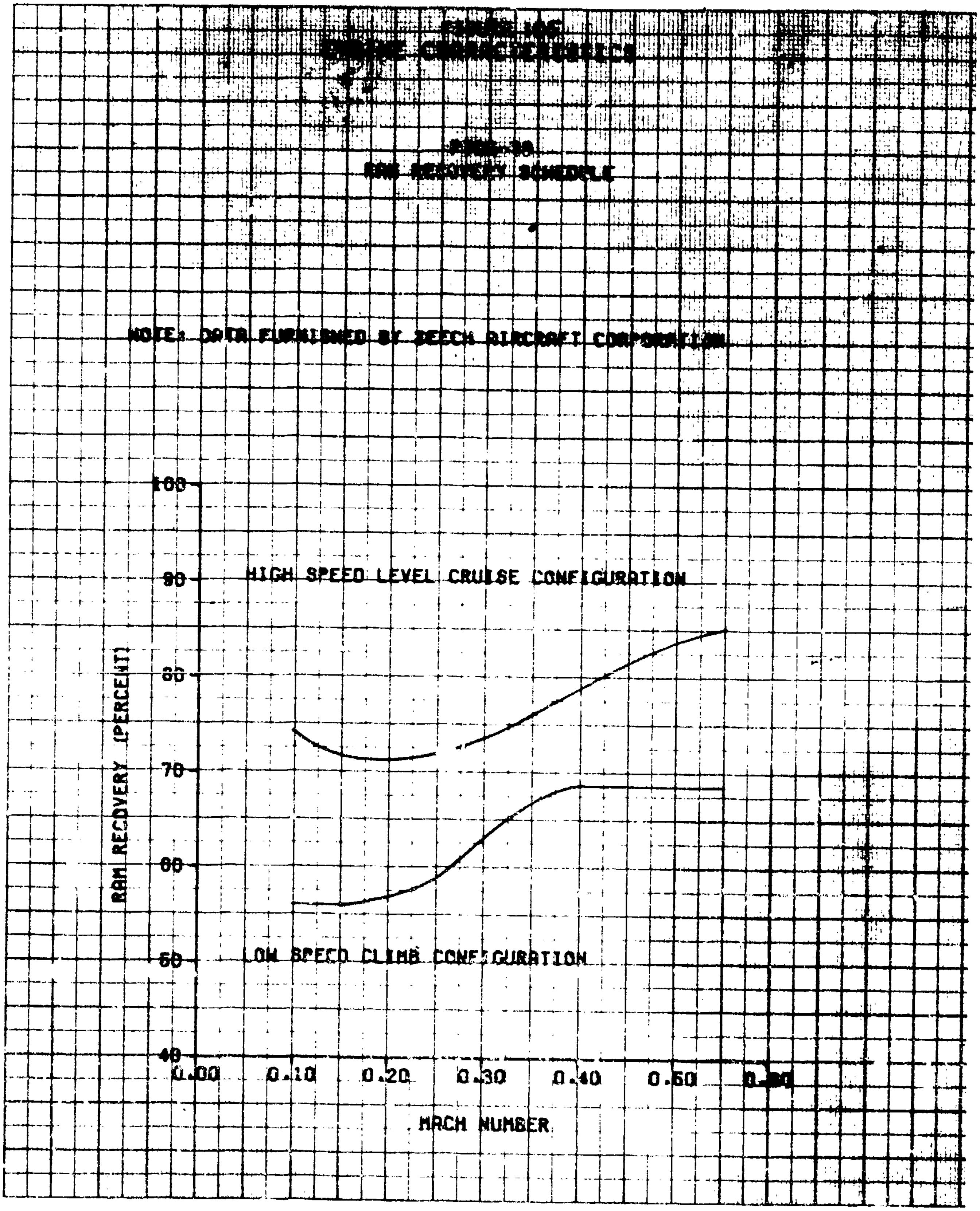


FIGURE 104
ENGINE CHARACTERISTICS
ENGINE MODEL PT6A-38
SPECIFICATION ENGINE BASED ON UACL COMPUTER DATA
MAXIMUM TIME BEFORE OVERHAUL
STANDARD DAY CONDITIONS

NOTE: 1. ACCESSORY LOSSES: 10.0 SHP AT OR BELOW 10° C, 11.5 SHP ABOVE 10° C
2. BLEED AIR: 6.75 LB/MIN AT OR BELOW -1.1° C, 5.5 LB/MIN ABOVE 1.1° C
3. ENGINE INLET PRESSURE RECOVERY SCHEDULE FIGURE
4. INLET TEMPERATURE RISE $T_{t2} = TA(1+.2M^2)$
5. PROPELLOR SPEED = 1800 RPM




RAM RECOVERY SCHEDULE
NOTE: DATA FURNISHED BY BEECH AIRCRAFT CORPORATION
HIGH SPEED LEVEL CRUISE CONFIGURATION
LOW SPEED CLIMB CONFIGURATION
RAM RECOVERY (PERCENT)
100
90
80
70
60
50
40
0.00
0.10
0.20
0.30
0.40
0.50
0.60
MACH NUMBER

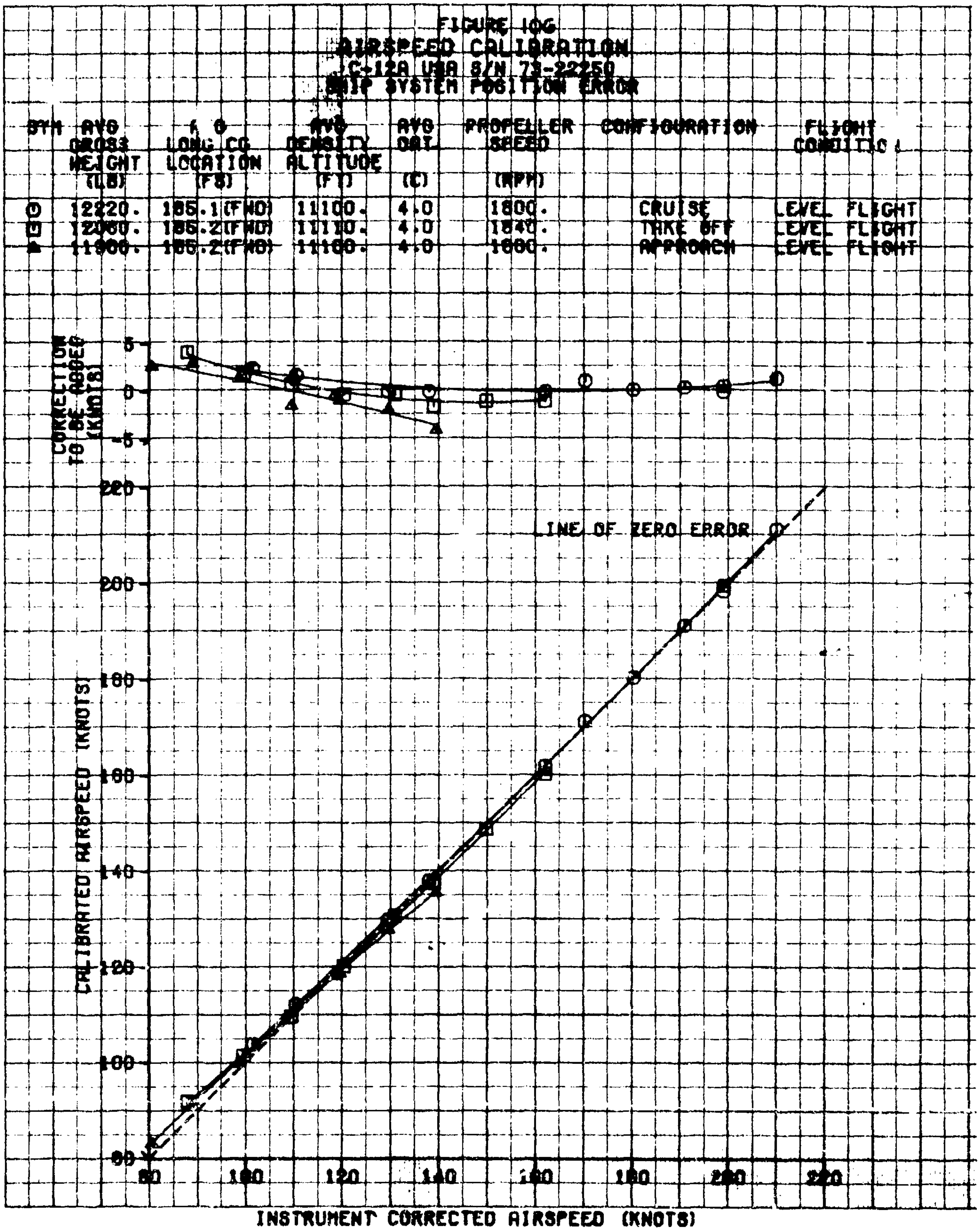


FIGURE 106
AIRSPEED CALIBRATION
C-12A USA S/N 73-22250
SHIP SYSTEM POSITION ERROR
SYM AVG GROSS WEIGHT (LB)
A G LONG CG LOCATION (FS)
AVG DENSITY ALTITUDE (FT)
AVG OAT (C)
PROPELLER SPEED (RPM)
CONFIGURATION
FLIGHT CONDITION
12220. 185.1(FWD) 11100. 4.0 1800. CRUISE LEVEL FLIGHT
12060. 185.2(FWD) 11110. 4.0 1840. TAKE OFF LEVEL FLIGHT
11900. 185.2(FWD) 11160. 4.0 1600. APPROACH LEVEL FLIGHT
CORRECTION TO BE ADDED (KNOTS)
LINE OF ZERO ERROR
CALIBRATED AIRSPEED (KNOTS)
INSTRUMENT CORRECTED AIRSPEED (KNOTS)

APPENDIX H. DEFINITIONS, ABBREVIATIONS, AND SYMBOLS

This list includes most of the symbols used in this report. However, certain portions of the report use special or unusual abbreviations and symbols. The meaning of these is made clear in the text of the report and, when that is the case, the abbreviation or symbol will not be found in this list. Also, certain symbols have more than one meaning; however, the context should make the meaning clear.

Symbols and Abbreviations	Definition	Unit
ANA	Air Force Navy Aeronautical	--
AC	Alternating current	--
b	Wing span	feet
C_{D_o}	Minimum coefficient of drag of the propeller-feathered drag polar	--
C_D	Coefficient of drag	--
$C_{D_{BL}}$	Base-line coefficient of drag	--
$C_{D_{PF}}$	Powered flight coefficient of drag	--
C_P	Coefficient of power	--
C_L	Coefficient of lift	--
Cont	Continuous	--
D	Drag	--
De	Degree	°C
e	Oswald's span efficiency factor	--
f	Equivalent flat plate area	ft^2
F_N	Jet thrust	pounds
g	Acceleration of gravity	ft/sec^2
H_D	Density altitude	feet

Hp_i	Indicated pressure altitude	feet
Hp	Pressure altitude	feet
Hp_{ic}	Instrument corrected pressure altitude	feet
J	Advance ratio	--
L	Lift	pounds
MAC	Mean aerodynamic chord	--
Max	Maximum	--
MCP	Maximum continuous power	--
Min	Minimum, minute	--
N_P	Propeller speed	rpm
N_1	Gas producer speed	percent
$N_1/\sqrt{\theta}$	Referred shaft horsepower	--
N_2	Power turbine speed	rpm
NAMPP	Nautical air miles per pound of fuel	--
NU	Nose up	--
ND	Nose down	--
OAT	Outside air temperature	°C
p	Roll rate	radians/sec
P_a	Ambient pressure	in. of mercury
P_o	Standard-day, sea-level pressure	in. of mercury
psi	Pounds per square inch	lb/in.^{2}
q	Dynamic pressure	lb/ft^2
Q	Torque	ft-lb

ref	Referred, reference	--
R/C	Rate of climb	ft/min
S	Wing area	ft^2
SE	Single engine	--
SHP	Shaft horsepower	--
$SHP/\delta\sqrt{\theta}$	Referred shaft horsepower	--
SL	Sea level	--
S/N	Serial number	--
STD	Standard	--
T_a	Ambient air temperature	°C
T_C'	Coefficient of thrust	--
T_i	Indicated air temperature	°C
T	Thrust	lb
T_{ic}	Instrument corrected air temperature	°C
THP	Thrust horsepower	HP
T_o	Sea-level, standard-day static temperature	°K
UHF	Ultra high frequency	--
V_{cal}	Calibrated airspeed	knot
VHF	Very high frequency	--
V_i	Indicated airspeed	knot
V_{ic}	Instrument corrected airspeed	knot
V_T	True airspeed	knot
V_{MC}	Airspeed for minimum control	knot
V_S	Stall airspeed	knot

V_H	Maximum airspeed for level flight	knot
V_{MO}	Maximum operating airspeed	knot
V	True airspeed	ft/sec
W_a	Engine airflow	lb/hr
W	Weight	pounds
°C	Degrees Centigrade	degrees
°F	Degrees Fahrenheit	degrees
°K	Degrees Kelvin	degrees
Δ	Difference	--
$\Delta C_{D_{PF-BL}}$	Difference in coefficient of drag due to thrust effect	--
ΔV_{PC}	Airspeed position error correction	--
ζ	Damping ratio	--
θ	Temperature ratio, descent angle	degrees
δ	Pressure ratio	--
σ	Density ratio	--
ρ	Air mass density	slug/sec^3
ω_d	Damped natural frequency	radians/sec
ω_n	Undamped natural frequency	radians/sec
a	Angle of attack	degrees
ϕ	Roll or bank angle	degrees
η_p	Propeller efficiency	--
ϕ/β	Roll-to-yaw ratio	--

dh/dt	Tapeline rate of descent	ft/min
π	3.14159	--
η_{in}	Inlet duct efficiency	percent

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US Naval Air Test Center	1
Hq US Coast Guard	1
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US Air Force Flight Dynamics Laboratory (TST/Library)	1
US Air Force Flight Test Center (SSD/Technical Library, DOEE)	3
US Air Force Electronic Warfare Center (SURP)	1
Department of Transportation Library	1
FAA, Wichita Engineering and Manufacturing District Office	1
Beech Aircraft Corporation	5
United Aircraft of Canada Ltd	5
Defense Documentation Center	12

United States Military Mission with the Iranian Army	2
Joint United States Military Aid Group to Greece	1
Joint United States Military Mission for Aid to Turkey	1
Office of Defense Representative, Pakistan	1
Hq US Readiness Command	1
US Army Engineer Division, Mediterraean	3
US Army Recruiting Command	1
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